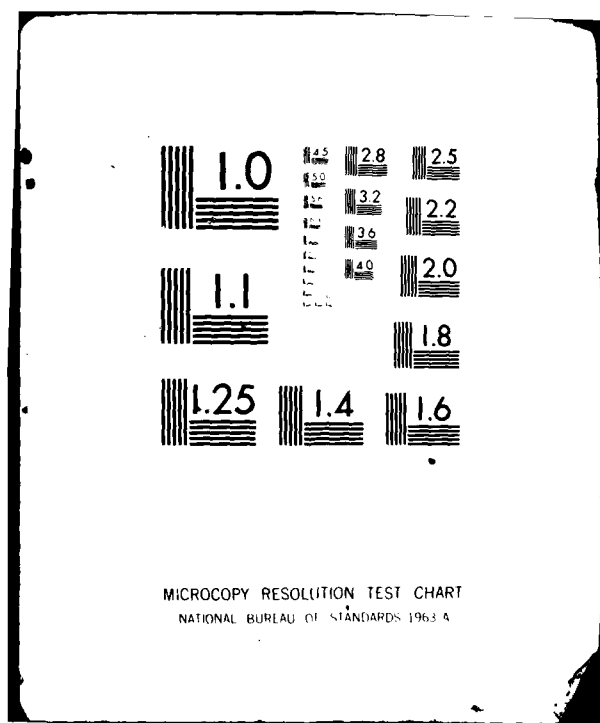


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PROGRESS TOWARDS A THEORY OF JET FLAP THRUST RECOVERY

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Rockwell International
North American Aircraft Operations
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the thrust coefficient increases. It is concluded that the loss of recovery is due to the jet drag for values of the thrust coefficient less than unity; for larger values, the loss is increased by flow separation from the airfoil.

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I. INTRODUCTION

A jet sheet directed downwards from the trailing edge of a wing, as shown in Figure 1, significantly increases the wing lift, especially at low speeds. Lift is generated both as a direct reaction to the jet impulse and as a result of changes in the wing surface pressure distribution induced by deflection of the main stream. This phenomenon is called the jet flap effect by analogy to the action of a mechanical flap. It makes some contribution to the forces induced by all powered lift systems.

In addition to generating a total lift on the wing greater than the vertical component of the jet impulse, the jet flap also induces a thrust greater than the horizontal component of the jet impulse. This was discovered during testing of jet flapped airfoils at the British

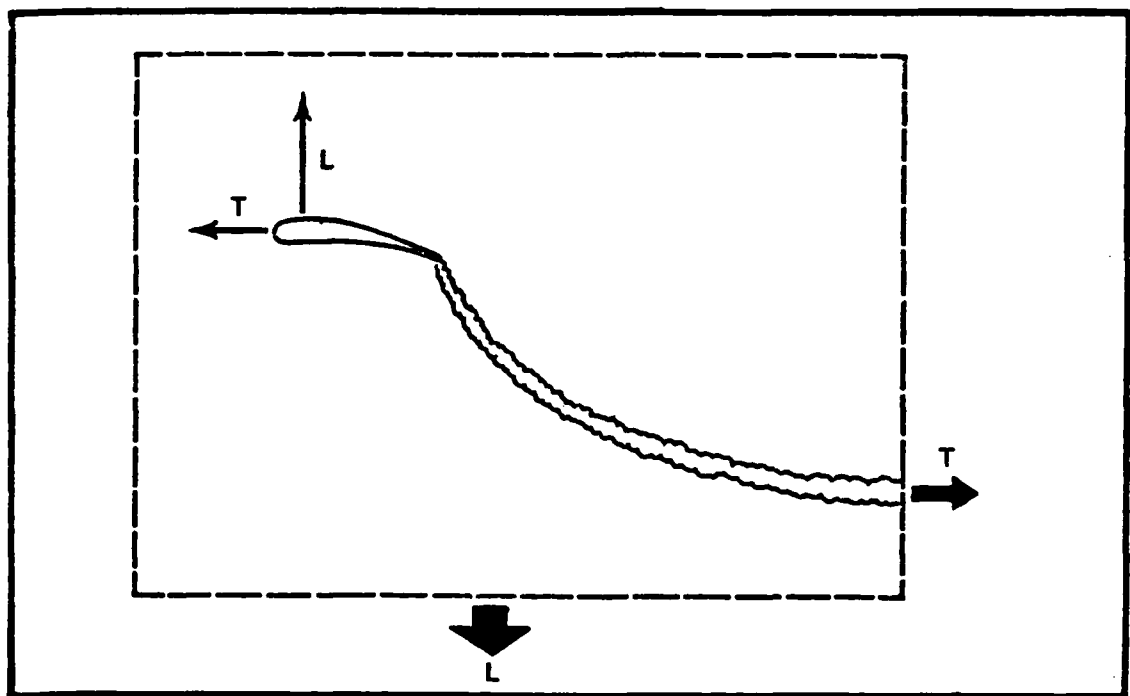


Figure 1. Jet Flap Thrust Recovery Hypothesis

National Gas Turbine Establishment. As a result of this discovery, Stratford (ref 1) suggested the hypothesis that, in ideal flow, the total jet thrust is always recovered as a horizontal force, regardless of the initial deflection angle, because the jet is ultimately turned in the direction of the free stream. He did not attempt a formal proof of this hypothesis, but reasoned from first principles that the force on the airfoil must be equal but opposite to the momentum of the deflected jet.

This thrust recovery hypothesis was received with considerable skepticism, but the existence of the phenomenon was established by tests performed at Stanford University, when Foley (ref 2) measured almost complete thrust recovery for jet deflection angles, θ , up to 60° . However, Stratford (ref 3) and Williams, et al (ref 4) measured only partial recovery at all deflection angles, with large decreases in recovery as the jet deflection angle increased to 90° . In subsequent tests with Foley's model, Tsongas (ref 5) also found a decrease in recovery when the deflection angle was increased beyond 60° , although Hynes (ref 6) found that the flow separated from the tunnel wall in this case.

The loss in thrust recovery has been attributed to various three-dimensional and real fluid effects: Maskell and Spence (ref 7) pointed out that deflection of the jet wake by the downwash behind a finite wing would reduce the recovery, even in ideal flow. Stratford (ref 3) and Wygnanski (ref 8) reasoned that the thrust recovery is decreased in real fluids by the jet entrainment, which induces a form drag on the airfoil; they suggested that the entrainment and, therefore, the jet drag increases with the jet deflection angle. More recently, Wilson (ref 9) used an empirical entrainment function to compute the jet drag and show that it is on the order of the observed loss in recovery. On the other hand, Williams, et al (ref 4) argued that flow separation on the airfoil would also cause a loss of thrust, while Tsongas (ref 5) suggested that separation of the main stream from the jet flap was the loss mechanism. Thus, while there is clear evidence of a thrust recovery phenomenon, the real fluid effects which reduce the recovery are incompletely understood.

Various analytic methods have been developed to predict the increments of aerodynamic lift and pitching moment induced by the jet flap. These methods are based on Spence's (ref 10) now classical jet flap theory, in which the inertia of the jet sheet is related to the strength of an equivalent vortex sheet. The original theory has been extended to include, for example, three dimensional effects (Maskell and Spence, ref 7), ground effects, (Lissaman, ref 11), and jet oscillations (Simmons and Platzner, ref 12). In addition, the theory has been applied to over-wing-blowing (Lan, ref 13) and ejector wing (Wilson, ref 9) configurations.

Although good predictions of the lift and moment have been obtained, none of these methods predict the loss of thrust recovery. In current practice, complete recovery is assumed for calculating lift, while an empirical function derived from the Stanford data is used to specify a reduction in the thrust recovery. Entrainment functions have been useful for estimating the magnitude of the thrust loss, but such functions treat just half of the jet/airfoil interaction, the jet drag induced on the airfoil; the other half of the interaction, the equal but opposite reduction of the jet thrust, is neglected. Since the reduction in thrust affects the computed position of the jet, which in turn affects the computed force on the airfoil, a method for predicting the effect of the jet flap should include this interaction.

The jet flap methods are further limited by linearizing assumptions to low values of the jet thrust coefficient and small airfoil and jet deflection angles. Thus, they cannot be used to predict the takeoff or landing performance of V/STOL aircraft for which current interest in the jet flap effect is greatest. Addessio and Skiftad (ref 14) recently addressed some of these low speed effects, but made the usual assumption of complete thrust recovery.

Such semi-empirical methods are only useful for small variations from the original data base. An essentially complete range of jet deflection

angles has been tested, but all of the data are for jet thrust coefficients less than unity. Since the thrust coefficient is much greater than unity for most of the V/STOL flight regime, there are no data for either the force on the airfoil or the trajectory of the jet sheet in this case.

The purpose of this report is to describe progress in developing a theory of jet flap thrust recovery at the low speeds and large deflection angles characteristic of V/STOL lift systems. A combination of analysis and testing has been utilized to develop this theory. A viscous/inviscid interaction analysis of the jet flap has been devised to compute the contribution of the jet drag to the reduction in thrust recovery. In addition, wake survey and airfoil surface pressure measurements have been made for a two-dimensional, jet flapped airfoil over a range of jet thrust coefficients. In the next section of this report, the mechanism of thrust recovery and the way in which entrainment and separation reduce the recovery are discussed. The jet drag analysis is then presented in the following section. The experimental apparatus and test procedure are described in the section after that. In the last section, the analytical results are compared to the measured thrust recovery. It is concluded that the thrust recovery is nearly complete for small values of the jet deflection angle; but for larger angles, the recovery decreases as the thrust coefficient increases. For values of the thrust coefficient on the order of unity, the thrust loss is due to jet drag; for larger values, the loss is increased by flow separation from the airfoil.

II. JET FLAP THEORY

MECHANISM OF THRUST RECOVERY

The thrust recovery is generated by equal but opposite pressure forces induced on the airfoil and jet sheet. According to Spence's (ref 10) theory, the inertial forces which resist deflection of the jet sheet are balanced by a pressure difference in the external flow along the upper and lower boundaries of the jet. This pressure difference can be related to the strength of an equivalent vortex sheet located along the jet axis. Then, in mathematical terms, the vorticity in a section of the jet sheet induces an upwash velocity of magnitude, $\Gamma_j/2\pi r$, on a vortex segment bound in the wing, as shown in Figure 2. An upstream force, $\rho \Gamma_w(\Gamma_j/2\pi r)$, is therefore induced on the wing. Similarly, the wing vortex induces a downwash, $\Gamma_w/2\pi r$ on the jet vortex which then experiences an equal force $\rho \Gamma_j(\Gamma_w/2\pi r)$ directed downstream.

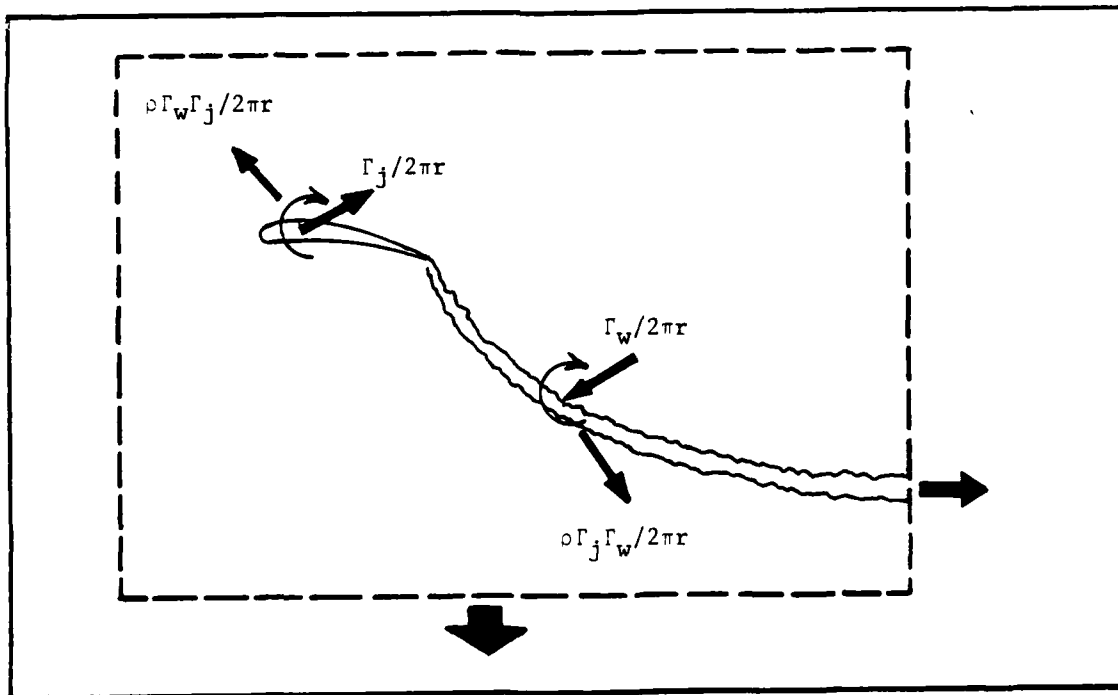


Figure 2. Mutually Induced Forces on a Vortex Pair

It is the integrated effect of all such vortex pairs which results in the apparent "thrust recovery." Figure 3 is a sketch of the forces on the airfoil, jet, and a surrounding control volume. As seen in the figure, the jet is not actually deflected, as by a vane. Rather, the interaction between the airfoil and jet induces a horizontal force, $\rho w \gamma$ on the jet and a corresponding reaction force on the airfoil. Simultaneously, the interaction of the main stream with the jet cancels the vertical component of the jet thrust and produces an equal reaction on the main stream. Thus, the horizontal thrust of the jet is increased, without any reduction in the vertical reaction force on the airfoil.

If the entrainment of the jet is neglected, Spence's theory predicts recovery of the total jet thrust as a horizontal force. This can be shown from the equilibrium of the forces which then act on the jet. Consider the segment of the jet shown in Figure 4. The forces due to the main stream and the wing vortex system deflect the jet through an angle, θ ,

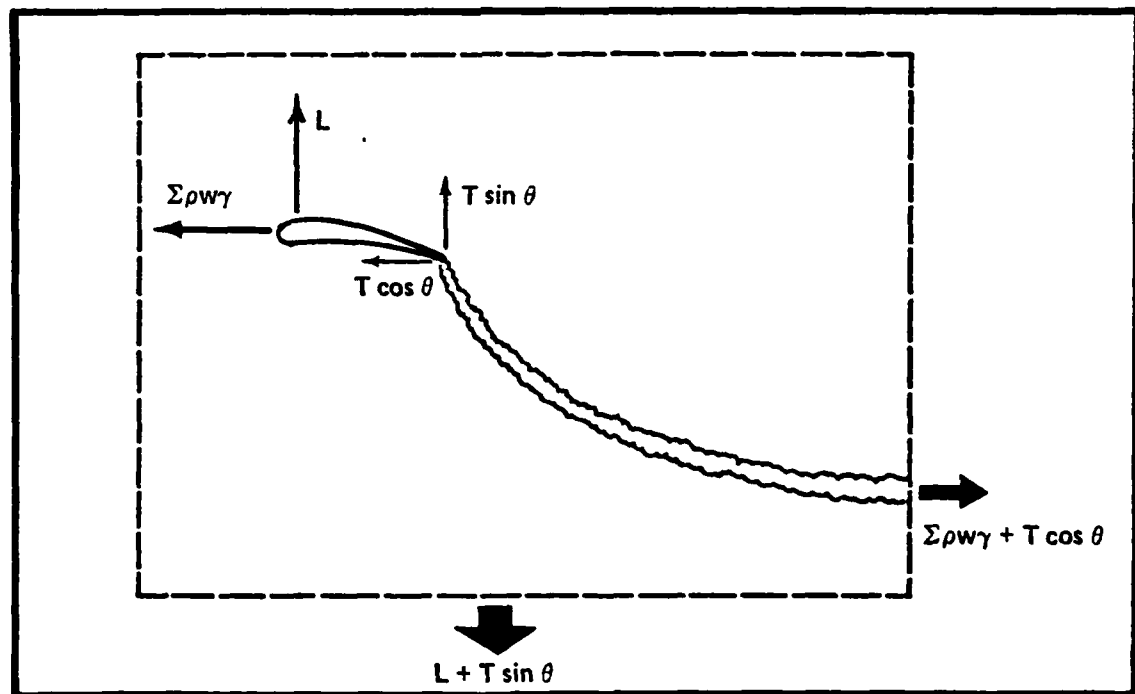


Figure 3. Forces Induced on an Airfoil and Jet Flap

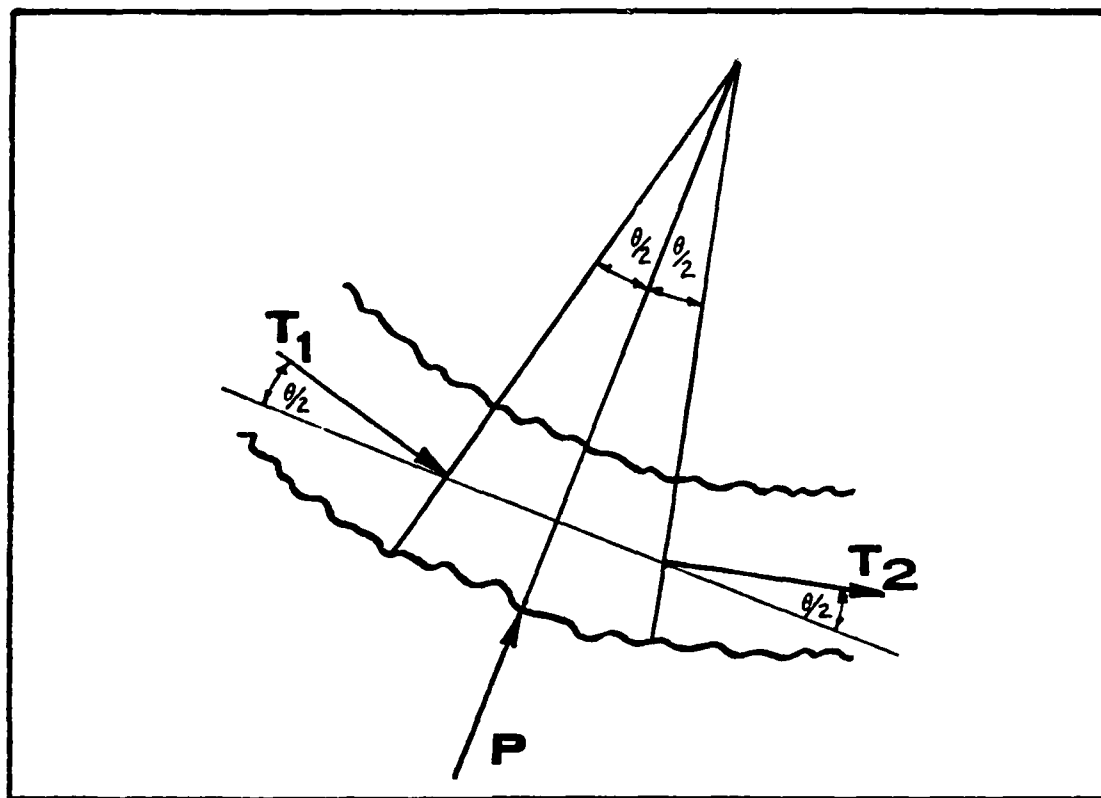


Figure 4. Forces on a Section of the Jet Sheet

as shown. The initial thrust of the jet is T_1 and the final thrust is T_2 . Since entrainment has been neglected, the jet will be a streamline of the flow and the pressure force, P , must therefore act normal to the segment.

The summation of forces in the direction of T_2 yields

$$T_2 = T_1 \cos \theta + P \sin \frac{\theta}{2} \quad 2-1$$

The summation of forces normal to the direction of T_2 gives

$$P \cos \frac{\theta}{2} = T_1 \sin \theta \quad 2-2$$

Solving this equation for P and substituting in Equation 2-1 gives $T_2 = T_1$. Therefore, the thrust of the jet is conserved. Because the jet consists of a series of these segments, its total thrust is ultimately recovered in the far wake as a horizontal force.

The horizontal force induced on the jet is the integral of the corresponding pressure difference across the jet; that is,

$$\Sigma p w y = \int_0^{\infty} dP \sin \theta d\theta \quad 2-3$$

According to Spence (ref 10), $dP = T/R$. Since T is constant and $ds = R d\theta$, integration yields

$$\Sigma p w y = T(1 - \cos \theta) \quad 2-4$$

The sum of this force and the horizontal component of the initial jet momentum gives the total jet thrust. The vertical force on the jet is similarly found to be $T \sin \theta$.

This thrust recovery mechanism was not verified by the Stanford experiments: Quambeck (ref 15) integrated the surface pressure distributions as the jet deflection angle was varied and, finding that the horizontal force did not change, he concluded that the entire jet thrust was felt as an internal pressure on the nozzle. However, in these tests, the jet was deflected by utilizing the Coanda effect (Metral and Zerner, ref 16) to turn it over a short flap. The reaction to the force which deflects the jet over the flap appears as a reduction in the static pressures on the Coanda surface. The resulting pressure drag is equal

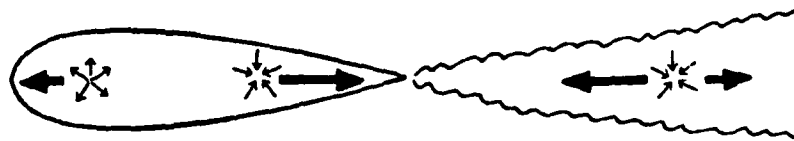
but opposite to the thrust induced by the jet flap because the force initially required to deflect the jet over the flap must be the same as the force necessary to bend it back in the direction of the free stream. As a result, the net horizontal force is independent of the jet deflection angle. Today, it is no longer doubted that the thrust recovery is induced on the airfoil surface, primarily as a leading edge suction.

THRUST LOSS HYPOTHESES

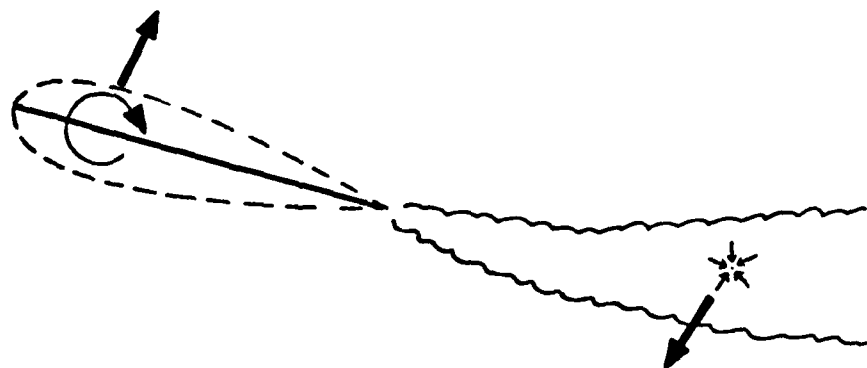
Real fluid effects reduce the thrust recovery through two separate mechanisms. The very large suction peak generated near the leading edge of the airfoil increases the possibility of flow separation in this region. If the flow does separate, the resulting loss of suction reduces the thrust on the airfoil, while entrainment of the low momentum wake fluid by the jet causes a corresponding loss of thrust. Separation becomes more likely as the jet thrust coefficient and deflection angle are increased.

However, even if the flow does not separate from the airfoil, entrainment reduces the thrust recovery. A jet drag is induced by the mutually induced forces on the vortex sheet bound in the wing and a sink distribution which represents the entrainment of the jet. The origin of the force on the wing is relatively easy to understand; the jet sinks change the surface pressure distribution so as to increase the wing drag. There are two contributions to this force, as shown in Figure 5. The jet drag due to thickness originates in the interaction between the jet sinks and the source distribution which defines the airfoil thickness. The jet drag due to lift originates in the interaction between the jet sinks and vortex distribution which determines the wing lift.

The origin of the reaction force on the jet may not be as apparent. This force is conceptually similar to the drag experienced by a sink



SINK-SINK DRAG DUE TO MIXING



VORTEX-SINK DRAG DUE TO MIXING

Figure 5. Origin of the Mixing Loss

in an external stream. However, the sink/vortex interaction is only an irrotational simulation of the jet mechanism. The fluid entrained into the jet becomes rotational, so that an understanding of how the reaction force actually develops requires consideration of the jet mixing process.

This process is basically an inelastic collision between the jet and surrounding fluid. As such, jet mixing is governed by the same laws of momentum and energy conservation as simple collisions between discrete particles. If the mixing occurs in a region of constant static pressure, the thrust of the jet is conserved. But if the jet passes through

a region where the static pressure, P^* , is different from the undistributed pressure at infinity, P_∞ , the mixing changes the momentum flux. The following simple analysis illustrates this phenomenon.

Inside the region where the pressure is P^* , the velocities in the jet and coflowing stream, before mixing, are assumed to be

$$U_j^* = \left[\frac{2(P_0 - P^*)}{\rho} \right]^{1/2} \quad 2-5$$

and

$$U_s^* = \left[U_\infty^2 + \frac{2(P_\infty - P^*)}{\rho} \right]^{1/2} \quad 2-6$$

in which P_0 is the stagnation pressure of the jet and U_∞ is the velocity of the coflowing stream at infinity. Momentum is conserved during the mixing process itself, so that

$$\dot{m}_j U_j^* + \dot{m}_s U_s^* = (\dot{m}_j + \dot{m}_s) U_m^* \quad 2-7$$

in which \dot{m}_j is the initial mass flux of the jet, \dot{m}_s is the quantity of mass entrained by the jet, and U_m^* is the average velocity of the mixed flow.

The velocity of the mixed stream changes as it passes into the region of undisturbed pressure; the velocity becomes

$$U_m = \left[U_m^{*2} - \frac{2(P_\infty - P^*)}{\rho} \right]^{1/2} \quad 2-8$$

The ratio of the final thrust of the mixed stream, $(\dot{m}_j + \dot{m}_s) U_m$, to the thrust, $\dot{m}_j U_j$, obtained by an isentropic expansion of the jet, without

mixing, to the final pressure may be evaluated by substituting in turn for U_m , then U_m^* , and finally for U_j^* and U_s^* . Performing these substitutions yields for this ratio

$$\phi = \left[1 + 2M \left((1+H)^{1/2} (U^2+H)^{1/2} - H \right) + M^2 U^2 \right]^{1/2} - MU \quad 2-9$$

in which $M = \dot{m}_s / \dot{m}_p$, is the entrainment ratio, $U = U_\infty / U_j$ is the velocity ratio, and $H = 2(P_\infty - P^*) / U_j^2$ is the normalized pressure change.

If there is no mixing between the jet and the coflowing stream, then $M = 0$ and the thrust of the jet is conserved; that is, $\phi = 1$. Similarly, if there is no pressure change, then $H = 0$ and the solution reduces to that for free jet mixing; the thrust is conserved in this case, also. However, if the pressure in the mixing region is less than the pressure at infinity, then $H < 0$ and the thrust of the jet increases. For the case in which $U_\infty = 0$, the thrust ratio has the simple form,

$$\phi = (1 + M)^{1/2} \quad 2-10$$

and the jet thrust increases with increasing entrainment. This is the basic mechanism of ejector thrust augmentation, which has been discussed in greater detail by Bevilaqua (ref 17).

If the pressure in the mixing region is above ambient, the thrust of the jet is reduced. For the case in which $U_s^* = 0$, the thrust ratio also has a simple form

$$\phi = 1 - MU \quad 2-11$$

and in this case the jet thrust decreases with increasing entrainment. Because there is a region of increased static pressure behind the trailing edge of the wing, the mixing loss mechanism will reduce the

jet thrust. According to Newton's law of action and reaction, the net loss of thrust must be equal to the jet drag on the wing.

The thrust loss is specified by a thrust recovery coefficient, r , which is defined to be a fraction of the total thrust; that is,

$$rT \equiv S + T \cos \theta_0 \quad 2-12$$

in which $T \cos \theta_0$ is the horizontal component of the jet thrust within the nozzle, and S is the horizontal component of the airfoil surface pressure distribution. In ideal, two-dimensional flow $S = T(1 - \cos \theta_0)$ and so $r = 1$. If a Coanda flap is used to reflect the jet, the pressure drag due to the Coanda effect, $T(1 - \cos \theta_0)$, is included in S . Therefore, the recovery is defined as

$$rT \equiv S + T \quad 2-13$$

In this case, $S = 0$ for ideal, two-dimensional flow so that $r = 1$, again. The evaluation of the recovery coefficient will be discussed in greater detail in Section 4.1, on data reduction.

In order to determine the relative importance of separation and jet drag in reducing the thrust recovery, the jet drag will be computed for unseparated flow and compared to the drag actually obtained on a wind tunnel model under the same conditions. Differences between these cases will be studied to quantify the loss mechanisms. In the next section, the method of calculating the jet drag will be developed from the principles outlined in this section.

III. COMPUTATIONAL METHODS

The Navier-Stokes equations for steady fluid motion are elliptic, which means that the domain of influence of a point disturbance is the entire flow volume. That is, pressure and stress gradients transmit the effect of local disturbances to every other point in the flow. Thus, the forces induced on the airfoil depend on the jet position, while the jet position, in turn, depends upon the forces on the airfoil. The basis of our analysis is the division of the flow field into two regions: the region of turbulent mixing within the jet, and the inviscid region surrounding it. The equal but opposite forces on the airfoil and jet will be computed by matching a viscous solution for the jet to a potential flow solution for the airfoil and jet flap. The significant elliptic effects are then transmitted through the pressure field of the external flow.

For the viscous jet solution, a finite difference analysis utilizing a two-equation turbulence model was developed. The basic scheme was that devised by Patankar and Spalding (ref 18) for thin shear layers: the full Navier-Stokes equations are reduced to a simpler set by the assumption that there is a primary direction of flow (along the jet), and that the diffusion of fluid properties is negligible in that direction. The effects of curvature on the development of the jet will be treated as a perturbation of this basic scheme. A jet flap panel method, in which the thrust of the jet is not assumed to be constant, was used for the inviscid solution.

INVISCID JET ANALYSIS

In the inviscid program, the potential flow continuity equation takes the form of the Laplace equation in terms of the velocity potential, $\nabla^2\phi = 0$. The mathematical problem to be solved requires finding a velocity potential which is harmonic, and satisfies the boundary condition of uniform flow at infinity and flow tangent to the surface of the

airfoil and the centerline of the jet. Since the shape of the jet is initially unknown, Spence (ref 10) made the important contribution to solving this problem by relating the inertia force in the jet to the strength of an equivalent vortex sheet,

$$T/R = \rho U \gamma$$

3-1

In this equation T is the jet thrust (which Spence assumed constant), R is the local radius of jet curvature, and γ is the vortex strength per unit length along the jet.

This force balance becomes an additional, dynamic boundary condition at each point on the jet sheet. In the inviscid analysis, the jet thrust is considered to vary in a known way. This variation is determined from the viscous solution for the turbulent jet computed in the previous iteration. The inviscid solution is obtained by iterating between the jet shape and the jet vorticity. For each iteration, the problem reduces to solving a set of linear influence equations with unknown vortex strengths. The boundary condition on both the airfoil and jet is that of no net normal velocity at each panel control point. A Gaussian elimination solution of the simultaneous equation is used.

The airfoil is described by an input set of X and Y coordinate points. These points may be used as the vortex panel endpoints, or an increased number of panels may be calculated using cosine spacing on the upper and lower surfaces and nonlinear interpolation of the input points. An airfoil geometry subroutine performs the interpolations if required and computes the individual panel angles and control point (panel centroid) coordinates. If an angle of attack case is run, all geometry is rotated, and all displacements are included.

In the solution, the airfoil is represented by distributions of linearly varying vorticity. The paneling scheme begins at the airfoil

trailing edge and proceeds forward over the lower surface and back over the upper surface to the trailing edge. Each vortex distribution, with the exception of the first consists of a ramp-up (increasing from zero) vorticity over one airfoil panel and a ramp-down (decreasing to zero) vorticity over the next airfoil panel. The first vortex distribution uses a constant vorticity over the first airfoil panel and a ramp down over the second airfoil panel. (This distribution was used to better represent the near constant pressure in this region under conditions of large jet deflection and high jet momentum.) This paneling scheme gives a number of unknown vortex strengths which is one less than the number of airfoil panel control points, at which the boundary conditions were defined. This over-specified problem is solved by the method of uniform error, (Bristow, ref 19). Figure 6 illustrates the airfoil vorticity paneling scheme. The jet was modeled as a thin sheet of vorticity using the assumptions of Spence (ref 10). Within each iteration, the jet shape was assumed to be known. For the first iteration, the

Linear Vorticity Distribution

N Unknown Vortex Strengths,
 $N + 1$ Control Points

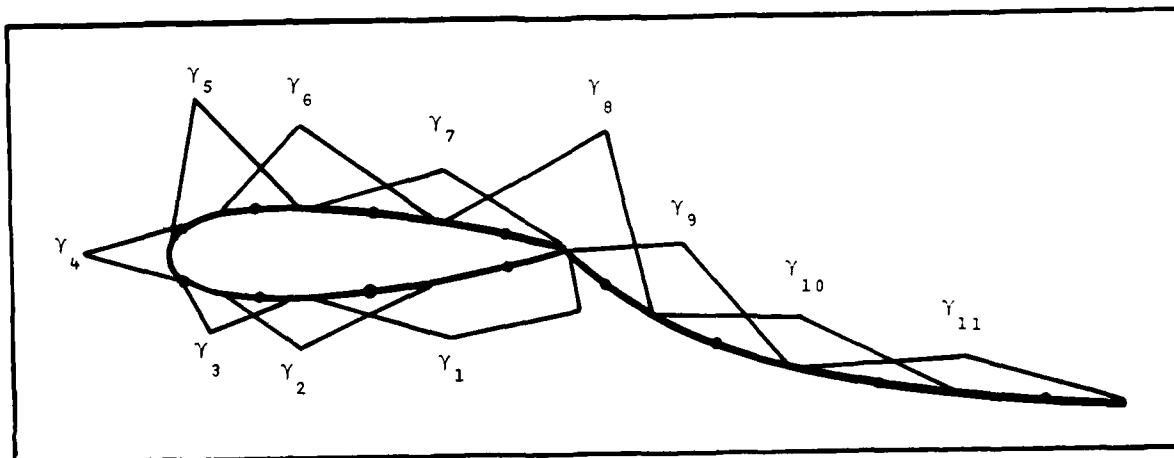


Figure 6. Vorticity Distributions on the Airfoil and Jet

empirical jet shape equation from Chandra (ref 20) was used. For subsequent iterations, the jet panel angles were computed using an iteration scheme described in a later section. Briefly, this scheme used the previous jet shape (panel angles) modified as guided by a new shape, computed from satisfying the dynamic jet boundary condition, Equation 3-1.

Jet panel lengths were set by a simple geometric progression factor, usually between 1.05 and 1.15. Thus, each panel length was larger than the one just upstream by the factor chosen. The first jet panel length was similarly related to the last airfoil upper surface panel length. This scheme had the distinct advantage of minimizing numerical problems associated with rapidly changing panel sizes. The jet panel lengths and angles were used to calculate the panel endpoint and control point (centroid) coordinates using a summation beginning at the airfoil trailing edge.

The jet was represented by distributions of linearly varying vorticity. The first jet vortex was a single ramp down function over the first panel. Subsequent vortex distributions were a ramp up over one panel and a ramp down over the next downstream panel. The second jet vortex distribution begins on the first jet panel. The last jet vortex distribution ramps down to zero strength at the trailing edge of the last jet panel.

When required, the jet entrainment effects are represented by a number of sink panels, coinciding with the jet panels, and a numerically integrated sink strength which decays as $X^{-1/2}$ from the jet trailing edge to a point far downstream (typically 1000 chord lengths). The sink strengths are computed from entrainment output from the viscous program and are modeled as constant over each jet panel. The initial value of the decaying sink function is set equal to the last panel's constant strength. The effect of these jet sinks is to modify the boundary conditions imposed at each control point. The jet sink panels are illustrated in Figure 7.

Constant Sink Distribution

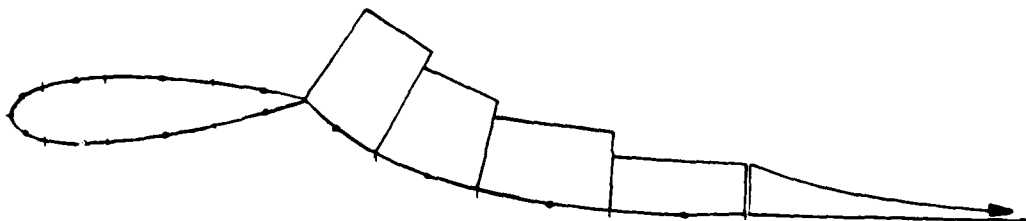


Figure 7. Sink Distributions on the Jet Panels

Vortex and Sink Influence Equations

The equations used to compute the velocities induced by each of the linearly varying singularities (vortices or sinks) used in the inviscid flow analysis are given below. These equations were developed by Dvorak and Woodward (ref 21). Note that each equation is written in a coordinate system with its origin at the left hand edge of the singularity panel and with its ordinate axis lying along the singularity. Routines within the inviscid program handle rotation and translation of an influenced point into the panel coordinate system, and resolve the induced velocities into velocities normal and tangential to the influenced panel control point. The terms in the equations are defined in the sketch of the singularity coordinate system sketched in Figure 8.

The velocities induced by a ramp up vortex or sink distribution are:

$$U_V = W_S = \frac{S}{2\pi} \left\{ \frac{z}{c} \log \frac{\sqrt{(x-c)^2 + z^2}}{\sqrt{x^2 + z^2}} - \frac{x}{c} \left[\tan^{-1} \frac{z}{x} - \tan^{-1} \frac{z}{x-c} \right] \right\} \quad 3-2$$

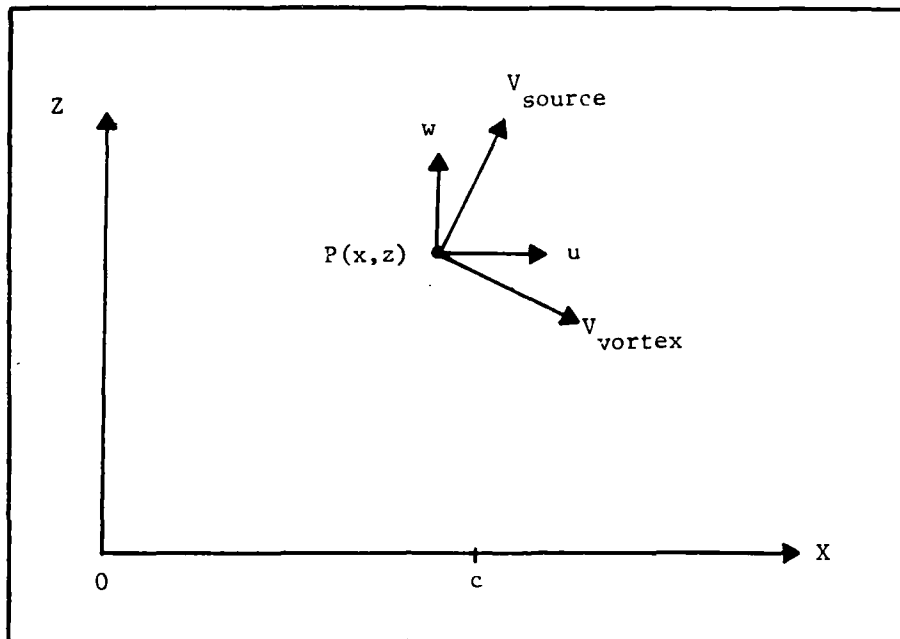


Figure 8. Singularity Coordinate System

$$-W_v = U_s = \frac{-S}{2\pi} \left\{ 1 + \frac{z}{c} \left[\tan^{-1} \frac{z}{x} - \tan^{-1} \frac{z}{x-c} \right] + \frac{x}{c} \log \frac{\sqrt{(x-c)^2 + z^2}}{\sqrt{x^2 + z^2}} \right\} \quad 3-3$$

The velocities induced by a constant vortex or sink distribution are:

$$U_v = W_s = \frac{S}{2\pi} \left[\tan^{-1} \frac{z}{x-c} - \tan^{-1} \frac{z}{x} \right] \quad 3-4$$

$$-W_v = U_s = \frac{-S}{2\pi} \log \sqrt{\frac{(x-c)^2 + z^2}{x^2 + z^2}} \quad 3-5$$

where s is the singularity strength, which is known for the sink distributions and set equal to V_∞ for the calculation of the vortex influence coefficients.

The velocities induced by a ramp down singularity distribution are calculated by subtracting the ramp up velocities from the constant singularity velocities. Velocities induced by the composite singularities (ramp up on one panel and ramp down on the next panel) are simply the sum of the two contributions computed separately.

Initial Jet Shape

While the initial jet shape is not critical with the iteration scheme used, a good first guess minimized computer run time. For the present inviscid flow program, the initial jet shape (in terms of panel angles) was obtained from the empirical jet angle function derived by Chandra (ref 20).

$$\frac{\theta}{\theta_0} = 0.64 \exp (-.82 s/C_\mu) + 0.36 \exp (-11.7 s/C_\mu) \quad 3-6$$

For any initial jet deflection, θ_0 , and jet momentum coefficient, C_μ , the jet angle at a given distance, s , along the jet arc could be computed. Jet panel lengths were computed as described above, and the initial panel angles were then the average of the angles at the panel endpoints. Initial jet position was computed from the individual panel angles and lengths.

Solution for Singularity Strengths

The airfoil and jet panel geometries and the influence equations, 3-2 to 3-5 are used to compute the influence coefficient matrix and the boundary condition vector. Terms in the influence coefficient matrix, A_{ij} , are the normal velocity induced by unit strength vortex distribution, i , at the control point, j . The terms in the boundary condition vector, B_j , are equal but opposite to the normal velocities which exist

at each control point due to the freestream velocity, V_∞ , and the jet sinks, if present. The following matrix equation results:

$$A_{ij} \gamma_j = B_i \quad 3-7$$

This system of equations is solved for γ_j by a Gaussian elimination procedure.

Pressure and Force Coefficients

The resulting pressure coefficients, C_{p_i} , are computed from the singularity strengths as follows:

On the airfoil:

$$C_p = \frac{P_{\text{local}} - P}{q} = 1 - V_T^2 \quad 3-8$$

where V_T is the total tangential velocity at the control point due to V_∞ and all singularities.

On the jet:

$$C_p = \frac{P_u - P_L}{q} = -2V_T \gamma_{\text{avg}} \quad 3-9$$

where γ_{avg} is the average of the vortex strengths at the panel endpoints.

Airfoil and jet normal force and axial force coefficients are computed by summing the product of the panel pressure coefficient and its respective projected area.

Calculation of "New" Panel Angles

"New" jet panel angle refers to the panel angles computed from the jet vorticity distribution and the dynamic boundary condition, Equation 3-1. They are different from the next iteration jet panel angles, which are computed from the last iteration angles by the iteration scheme described below. In terms of the local jet radius, R , and panel length, ΔS , the change in jet flow angle from one end of a panel to the other is

$$\Delta \theta = \Delta S / R \quad 3-10$$

but, from Equation 3-1, $R = T / \rho V \gamma$. Thus,

$$\Delta \theta = \Delta S \rho V \gamma / T \quad 3-11$$

In terms of the local pressure coefficient, $C_p = \rho V \gamma / q$ and jet thrust coefficient $C_\mu = T / qc$, the change in angle is

$$\Delta \theta = -C_p \Delta S / C_\mu C \quad 3-12$$

Each "new" jet panel angle is then the average of the panel endpoint angles found from the panel pressure coefficients, local momentum coefficient (input from the viscous program) and panel length.

Iteration Scheme

An iteration scheme was written which eliminates some of the problems associated with ordinary relaxation techniques. This scheme provides rapid convergence initially, is not as sensitive to oscillations near the jet trailing edge, and does not require the user to choose a weighting factor.

This iteration scheme does not use the "new" jet panel angles directly. Instead, they are used only as an indication of the direction of change. The magnitude of the change in panel angle is determined by the factor $1 - \frac{1}{2^n}$, where n is a positive integer which may be different for each panel. Initially, $n = 1$ and n is incremented by 1 each time the required direction of change is reversed. If the "new" panel angle is less than the angle used in the previous iteration, the previous angle is multiplied by the factor $1 - \frac{1}{2^n}$ using the current value of n . If the "new" panel angle is larger than the angle used in the previous iteration, the previous angle is divided by the factor $1 - \frac{1}{2^n}$. In either case, the next iteration jet panel angle is never allowed to change by an amount greater than "new" angle - "old" angle.

The iteration continues until a convergence criterion is met. The criterion used is that the average change in panel angle must be less than 0.01 times the initial jet deflection.

TURBULENT JET ANALYSIS

The sink strengths which represent the effect of the jet mixing are calculated from a viscous solution for the turbulent jet. It is possible to compute the mixing without including the elliptic effects by taking advantage of the flow geometry. Since there is a primary direction of flow (along the jet), the thin shear layer approximation can be applied. This reduces the governing elliptic equations to a parabolic set which can be solved by marching along the jet in the streamwise direction. By treating the radial pressure gradient as a perturbation of the axial pressure gradient, additional simplifications are achieved. These assumptions result in considerable savings in computer storage and running time compared to a solution of the complete elliptic problem.

The governing equations are derived from Reynolds equations for turbulent flow by neglecting streamwise diffusion and upstream convection. Thus, the effect of a local disturbance is transmitted downstream by pressure and convection and across the flow by pressure, convection and diffusion. In cylindrical coordinates, (r, θ) , the equations take the general form:

Conservation of Mass

$$\frac{\partial U}{r \partial \theta} + \frac{\partial V}{\partial r} + \frac{V}{r} = 0 \quad 3-13$$

Streamwise Momentum

$$U \frac{\partial U}{r \partial \theta} + V \frac{\partial U}{\partial r} + \frac{UV}{r} = - \frac{1}{\rho} \frac{\partial P}{r \partial \theta} - \frac{\overline{\partial u' v'}}{\partial r} - \frac{2 \overline{u' v'}}{r} \quad 3-14$$

Radial Momentum

$$U \frac{\partial V}{r \partial \theta} + V \frac{\partial V}{\partial r} - \frac{U^2}{r} = - \frac{1}{\rho} \frac{\partial}{\partial r} (P + p') \quad 3-15$$

in which U and V are the streamwise and radial velocity components. The radial perturbation pressure $p'(r, \theta)$ is decoupled from the mean streamwise pressure, $P(\theta)$.

These equations do not describe the effect of curvature on the flow. The "extra" terms originate in the choice of coordinate system and do not imply any additional physical processes. The Coriolis acceleration, $UV/r = \omega V$ and the centrifugal acceleration $U^2/r = \omega U$, arise from rotation of the velocity vector with respect to the coordinate system. Similarly, the extra stress term is a consequence of having chosen a cylindrical volume element; that is, since the sides of the element are not parallel,

the shear stress has a component, $\overline{u'v'}/r$, in the streamwise direction. Thus, the use of cylindrical coordinates is a convenience in this case, and the additional terms merely account for the difference between the actual accelerations of the fluid and those which appear to take place in the curvilinear coordinate system.

To facilitate solution, the governing equations are transformed into an ω - x coordinate system as shown in Figure 9. This provides the advantage of expanding the grid point spacing as the jet spreads. Thus, the maximum number of grid points are always contained within the jet, and there is no need to include additional, unnecessary grid points for the external stream. The stream function, ψ is defined as

$$\frac{\partial \psi}{\partial y} = \rho U$$

3-16

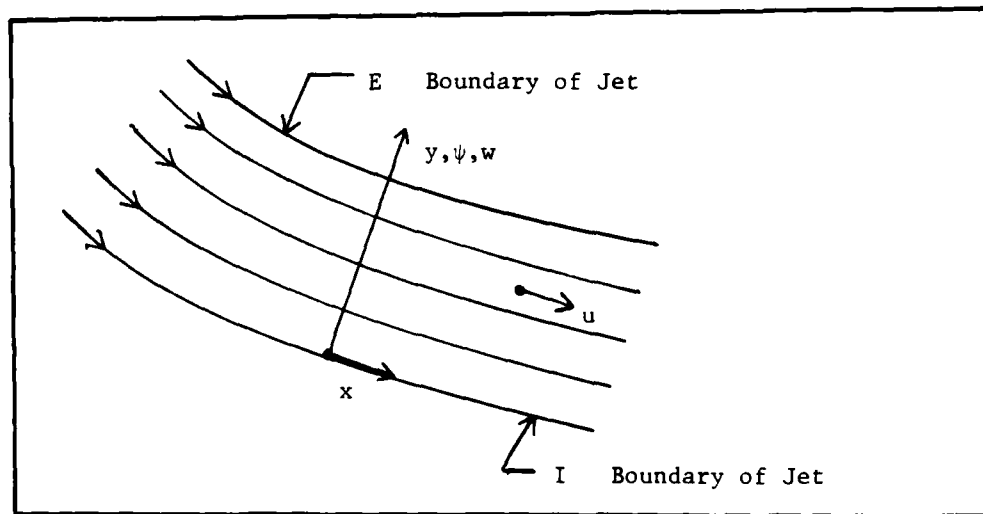


Figure 9. Transformed Coordinate System for Turbulent Jet

The variable, ω , is a dimensionless stream function defined by:

$$\omega = \frac{\psi - \psi_I}{\psi_E - \psi_I} \quad 3-17$$

The ψ coordinate lines are not true stream lines. They are used as a convenient method to handle flow in the cross stream direction.

Turbulence Equations

The effect of curvature is to increase the turbulent mixing rate. Recent proposals to model this effect have been based on the turbulence kinetic energy equation. According to the usual eddy viscosity assumption, the turbulent stress is first expressed in terms of a turbulent viscosity, μ_t , and the mean strain rates

$$\overline{u'v'} = \mu_t \left(\frac{\partial U}{\partial r} - \frac{U}{r} \right) \quad 3-18$$

Again, no new physical process is implied by the extra strain rate, U/r . Following Launder and Spalding (ref 22), the turbulent viscosity is assumed to depend on two parameters: the turbulence kinetic energy, k , and its rate of dissipation, ϵ . From dimensional arguments, the expression for the turbulent viscosity is

$$\mu_t = \frac{c_\mu \rho k^2}{\epsilon} \quad 3-19$$

in which c_μ is an empirical constant. A procedure for modifying the turbulence kinetic energy equations to account for the effect of curvature has been developed by Schum, et al. (ref 23). This method has been used here. Briefly, the effect of curvature is to transfer energy from the streamwise component of the turbulence to the transverse component.

Since it is the transverse component which produces most of the mixing in thin shear layers (Bradshaw, ref 24), this transfer of energy increases the mixing.

However, curvature has no effect on the total turbulence energy, because energy is simply transferred from one component to another. Thus, the turbulence kinetic energy equation cannot be used to calculate the effects of curvature. An additional production term was therefore added to the total turbulence energy equation to represent the increase of the transverse energy component by the curvature. In cylindrical coordinates, the equation for the turbulence energy becomes

$$\rho \frac{dk}{dt} = \overline{u'v'} \left(\frac{\partial U}{\partial r} - \frac{U}{r} \right) - \rho \epsilon + \frac{\partial}{\partial r} \left(\frac{\mu_t}{\sigma_k} \frac{\partial k}{\partial r} \right) + \overline{u'v'} \frac{2U}{r} \quad 3-20$$

in which the first term on the right hand side represents the usual production mechanism, the second term represents dissipation, the third term represents diffusion, and the last term represents the effect of curvature.

An equation derived by Chambers and Wilcox (ref 25) was used to compute ϵ , the dissipation of the turbulence energy. It contains the same curvature correction as the turbulence energy equation:

$$\rho \frac{d\epsilon}{dt} = (c_1 G - c_2 \rho \epsilon) \frac{\epsilon}{k} + \frac{\partial}{\partial r} \left(\frac{\mu_t}{\sigma_\epsilon} \frac{\partial \epsilon}{\partial r} \right) + \overline{u'v'} \frac{2U}{r} \quad 3-21$$

This turbulence model has five empirical constants: c_μ , c_1 , c_2 , σ_k , and σ_ϵ . The values of these constants recommended by Launder and Spalding (ref 22) were used for this analysis.

Finite Difference Equations

The governing mean flow and turbulence equations are solved simultaneously by iteration. In order to simplify the program logic, the momentum and turbulence equations are expressed in the general form:

$$\frac{\partial \phi}{\partial x} + (a + b\omega) \frac{\partial \phi}{\partial \omega} = \frac{\partial}{\partial \omega} \left(\Gamma \frac{\partial \phi}{\partial \omega} \right) + \frac{s}{\rho u} \quad 3-22$$

in which

$$a = \frac{\dot{m}_I}{\psi_E - \psi_I} \quad \text{and} \quad b = \frac{\dot{m}_E - \dot{m}_I}{\psi_E - \psi_I} \quad 3-23$$

The general dependent variable ϕ represents either U , V , k , or ϵ , while Γ and S represent the corresponding diffusion and source terms. Also, \dot{m} represents the mass flux across the jet boundaries shown in Figure 9. With all the equations expressed in this same form, the same solution procedure can be used.

A marching solution is performed in which the task is to obtain values of the flow variables at a downstream station when the values at the upstream station are known. With a given set of initial conditions, one can then make a series of forward steps in the streamwise direction to obtain the solution for the entire flow field. The flow field is modeled by specifying a number of control volumes across the flow in a direction normal to the mainstream direction, x , with grid points centered at the control volume faces upstream and downstream (Figure 10).

The method of specifying control volumes in the program is one where the control volume boundaries are input in the ω coordinate system (at $x = 0$) and the respective grid points are then located in the center of

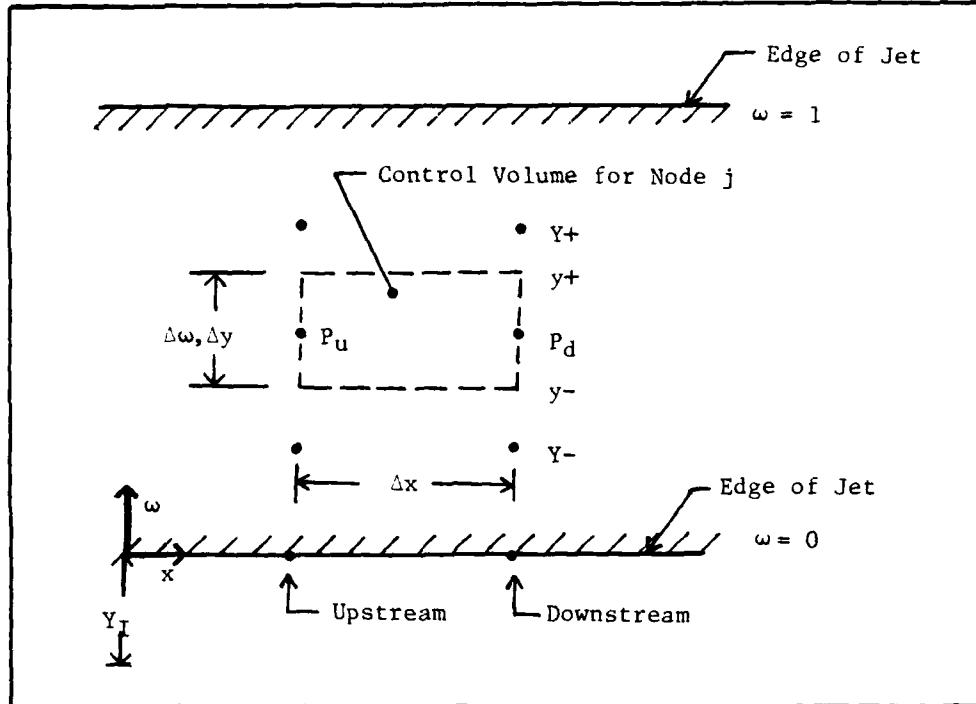


Figure 10. Control Volume and Coordinate System

each control volume (midway between y^+ and y^-). The points Y^+ and Y^- represent adjacent grid points, above and below grid point P_d . The values of Δy can change in the x direction, depending on the values of the U velocities in the control volume. The finite-difference equations are then formed by integrating the governing partial differential equations over the control volumes. In order to express the various terms in the equations in algebraic form, the following practices are used:

- 1) The value of the variable at the central grid point is assumed to prevail over the entire face of the control volume.
- 2) The downstream values are considered to prevail over the entire forward step.
- 3) The dependent variables are assumed to vary linearly between the grid points (this is modified when convection is large).

- 4) Strict compatibility is maintained between the neighboring control volumes, ie., the flux leaving a control volume through a face is made exactly equal to the flux entering the adjacent control volume through the same face.

The resulting algebraic equations (finite-difference equations) for each grid point have the following general form:

$$A_p \phi_p = A_{Y^+} \phi_{Y^+} + A_{Y^-} \phi_{Y^-} + A_{P_u} \phi_{P_u} + B \quad 3-24$$

where A_{Y^+} and A_{Y^-} represent the effect of cross stream convection and diffusion; A_{P_u} represents the upstream convection; B is the constant part of the source term

$$S = B + S_p \phi \quad 3-25$$

and A_p is defined by

$$A_p = A_{Y^+} + A_{Y^-} + A_{P_u} - S_p \quad 3-26$$

The symbolic term, ϕ , is used to represent the U, k, or ϵ parameters at grid locations, Y^+ , Y^- , u or d. Table I presents the corresponding source terms. The value of the A coefficients depend on which parameter is being considered. The coefficients A_{Y^+} and A_{Y^-} in Equation 3-24 are defined as:

$$A_{Y^+} = D_{Y^+}^* - \frac{1}{2} C_{Y^+} \quad 3-27$$

$$A_{Y^-} = D_{Y^-}^* + \frac{1}{2} C_{Y^-} \quad 3-28$$

D represents diffusion and C represents convection through the y^+ or y^- boundary. The subscripts for D and C indicate that these quantities are

Table I. Source Terms

ϕ	S_p	B
U	-	$-\frac{dP}{dx} - \frac{\partial}{\partial y} \left(\mu_t \frac{U}{r} \right) - \frac{dP'}{dx}$
k	$-\frac{\rho \epsilon}{k}$	$\mu_t \left(\frac{\partial U}{\partial y} - \frac{U}{r} \right)^2 - 2 \frac{U \mu_t}{r} \frac{\partial U}{\partial r}$
ϵ	$-\frac{c_2 \rho \epsilon}{k}$	$\rho c_1 c_{\mu} k \left(\frac{\partial U}{\partial y} - \frac{U}{r} \right)^2 - 2 \frac{U \mu_t}{r} \frac{\partial U}{\partial r} \frac{\epsilon}{k}$

There are two special things to note in the representations of the source terms

- (1) In the k equation, the $\rho \epsilon$ term has been reformed to $\left(\frac{\rho \epsilon}{k} \right) k^*$. This procedure improves the mathematical convergence accuracy.
- (2) In the ϵ equation, the $c_1 G$ term has been reformed by expressing the μ_t in G as $\frac{\rho C_D k^2}{\epsilon}$.

to be evaluated at the control volume boundary indicated by the subscript. D^* is the modified form of D depending on the size of C compared to D. This is called the "hybrid scheme" and represents a central difference scheme when C is less than twice D and an upwind difference scheme when C is twice D:

$$C = \{ m_I + \omega(m_E - m_I) \} \Delta X \quad 3-29$$

$$D = \frac{\Gamma \Delta X}{\Delta y'} \quad 3-30$$

$$\begin{aligned} \text{and } D^* &= D & \text{if } |C| < 2D \\ &= 1/2C & |C| \geq 2D \quad \text{and } C \text{ positive} \\ &= -1/2C & |C| \geq 2D \quad \text{and } C \text{ negative} \end{aligned}$$

where Γ , ω , and Δx are the values of the exchange coefficient, dimensionless stream function, and x stepsize, respectively, of the control volume boundary being considered. $\Delta y'$ is the distance between the two grid points on either side of the control volume boundary.

The coefficient A_{PU} represents the convection from upstream and is calculated by

$$A_{PU} = (\psi_E - \psi_I)_U \Delta \omega \quad 3-31$$

where $(\psi_E - \psi_I)_U$ is the total upstream flow rate and $\Delta \omega$ is the change in ω across the control volume being considered.

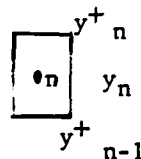
Once the solution of the differential equations is complete and values of the U velocities and ΔP have been calculated, the locations of the grid points and control volume boundaries in terms of y must be found. This is done by using the definitions of ψ and ω to obtain

$$dy = \frac{(\psi_E - \psi_I)}{\rho U} d\omega \quad 3-32$$

Now by using this expression and the known ω values that are input for the control volume boundaries the corresponding Δy can be found from

$$(\Delta y)_n = \frac{(\psi_E - \psi_I)}{(\rho U)_n} (\Delta \omega)_n \quad n = 2, 3, \dots, N \quad 3-33$$

for each n^{th} control volume. The y values of the control volume boundaries, y^+ , are then easily found by knowing the boundary value, y_I , (which can be set to 0) and the $(\Delta y)_n$ values:



$$y_n^+ = \frac{(\psi_E - \psi_I)}{(\rho U)_n} + y_{n-1}^+ \quad n = 2, 3, \dots, N+1 \quad 3-34$$

where $y_1^+ = y_I$ and y_{N+1}^+ will equal y_E .

The y location of the grid point n is calculated by using the centered grid point approach so

$$y_n = \frac{1}{2} (y_n^+ + y_{n-1}^+) \quad n = 2, 3, \dots, N=1 \quad 3-35$$

Pressure Equations for Curved Flow

In the viscous program, two pressure gradients are considered--one in the streamwise direction and the other in the cross stream direction, a result of jet curvature. With the addition of curvature effects, the change in the local pressure due to the centrifugal forces must be included in the mainstream momentum equation. However, in parabolic flow, these pressure changes must not change the total integrated mean pressure difference times area term for a cross section. In other words, only one value of mean pressure exists at a particular x station. The local pressure variation due to curvature is calculated by creating an additional pressure value, P' , from

$$\frac{\partial P'}{\partial y} = \frac{\rho U^2}{r} \quad 3-36$$

where r is the local curvature. Upstream values of the velocity, U_u , are used for downstream values since the latter have yet to be determined. With small Δx steps, as used here, this is a good approximation.

For the mean or global pressure to have only one value at a particular x station, it is necessary that:

$$\sum P'_j dA_j = 0 \quad j = 1, 2, \dots, N \quad 3-37$$

where N is the number of control volumes. The values of P'_j are used to determine a gradient dP'/dx for use in the source term in the U momentum equation (Table I).

Boundary Conditions

The viscous program requires boundary conditions to be specified at the inlet plane and along the jet surfaces. At the inlet plane (nozzle exit) a top hat velocity profile is used. The radial perturbation pressures are determined from Equation 3-36 in which the radius of curvature is specified by the inviscid program. The U velocity at the inlet is derived from a stream tube analysis, Figure 11. By combining the Bernoulli and continuity equations with the definition of C_μ , the initial jet velocity, $U_{J,i}$ can be obtained.

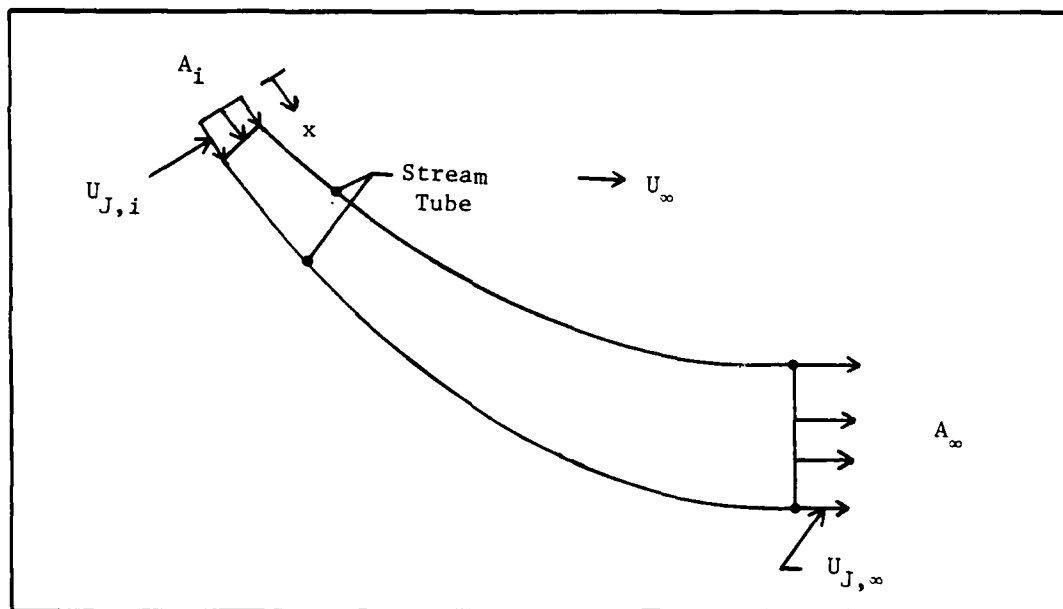


Figure 11. Stream Tube for the Deflected Jet

From Bernoulli

$$P_i + \frac{1}{2} \rho U_{J,i}^2 = P_\infty + \frac{1}{2} \rho U_{J,\infty}^2 \quad 3-38$$

and continuity

$$\rho U_{J,i} A_i = \rho U_{J,\infty} A_\infty \quad 3-39$$

It can be shown that $U_{J,i}$ can be obtained from the quadratic

$$(U_{J,i})^2 + \frac{2(P_i - P_\infty)U_{J,i}^2}{\rho} - \left(\frac{U_\infty^2 C_\mu C}{2 A_i} \right)^2 = 0 \quad 3-40$$

At any downstream distance, x , the velocity profile is dependent on the solution of the momentum equations with prescribed velocities as boundary conditions at the inner and outer surface of the jet. There are two ways to obtain the edge velocities, one is to use the velocities from the inviscid program and the other is to use the upstream radial pressures from the viscous program. The latter approach was used in the iteration procedure involving the inviscid and viscous programs. This serves as a check on the degree of convergence between the two programs since the velocities at corresponding locations should be the same when convergence is achieved. The inner and outer jet surface velocities are determined from:

$$(U_I)_{\text{Viscous}} = \sqrt{\frac{2}{\rho} (\bar{P} + P'_I)} \quad 3-41$$

$$(U_E)_{\text{Viscous}} = \sqrt{\frac{2}{\rho} (\bar{P} + P'_E)} \quad 3-42$$

in which P'_I and P'_E are determined from the previous upstream radial pressure distribution (Eq. 3-36), calculated in the viscous program. The mean

pressure, \bar{P} , is obtained from the inviscid velocities.

$$\bar{P} = \frac{1}{2} \rho \left(\frac{U_I + U_E}{2} \right)^2_{\text{Inviscid}} \quad 3-43$$

The velocity profiles in the jet are also dependent upon the entrainment at the inner and outer jet surface, \dot{m}_E and \dot{m}_I . The values are obtained from the relation given by Patankar and Spalding (ref 18)

$$\dot{m} = \left(\frac{\partial \mu_t}{\partial y} \right)_{\text{jet edge}} \quad 3-44$$

in which the viscosity is calculated in the solution of the momentum and turbulence equations. In some cases, step-to-step oscillations in the width of the jet can occur. One of the many approaches to minimize these oscillations (Patankar and Spalding, ref 18) is to utilize a multiplication factor, F , in the entrainment calculations, such that

$$\dot{m} = F \left(\frac{\partial \mu_t}{\partial y} \right)_{\text{jet edge}} \quad 3-45$$

where F is altered from one Δx step to the next by

$$F_{\text{new}} = F_{\text{old}} \left[\frac{|U_{N+1} - U_N|}{\Delta U^*} \right]^n \quad 3-46$$

where $|U_{N+1} - U_N|$ is the absolute value of the difference in velocity between the boundary and adjacent grid point, and ΔU^* is the velocity difference specified by the user as the minimum to which he wants the difference between U_{N+1} and U_N to fall. The exponent controls the rate of change of thickness if oscillations are a problem. Setting it equal to 0.05 has generally worked well. ΔU^* is generally specified as a

percentage of the maximum change in velocity in the velocity profile, say one or two percent of $U_{\max} - U_{\min}$. Even though the value of F may vary from step to step, on the average, the correct entrainment is calculated.

Solution of Finite Difference Equations

For each grid node (excluding boundary nodes) there is a corresponding finite difference equation (Equation 3-24) representing the momentum and turbulence equations. If there are M nodes used, there are $(M - 2)$ equations to be solved. These finite-difference equations are solved by using the tri-diagonal matrix algorithm having the generalized format:

$$A_j \phi_j = B_j \phi_{j+1} + C\phi_{j-1} + D_j^* \quad 3-47$$

for $j = 2, 3, \dots, M-1$. ϕ_1 and ϕ_M are prescribed boundary conditions, such as the jet edge velocities. Equation 3-47 represents Equation 3-24 in which the last two terms have been combined into the single value, D_j^* .

When calculating the coefficients for the equations, several values of flow variables and distances are needed. Upstream values are used when downstream values are not available. No iterations per forward step are performed. These iterations could improve the solution; however, the improvement is insufficient to warrant the increased work for two-dimensional parabolic problems. This lack of iteration is only a problem if the forward steps become too large and one is covering an area where the flow variables change a lot over one step. For the cases studied here, the forward Δx step was only 0.0005 ft over a total jet length of about 10 ft. Changes in the flow variables over such a small distance as 0.0005 ft are relatively small.

The only readily available data for a turbulent jet in a coflowing stream with a streamwise pressure gradient is that of Fekete (ref 26). Tests were conducted so as to provide self preservation. According to Newman (ref 27), self preservation is possible only if:

$$\frac{U_0}{U_\infty} = \text{constant} \quad 3-48$$

and $U_\infty \propto X^m \quad 3-49$

where

$$m = - \frac{\sqrt{2 + U_0/U_\infty}}{3\sqrt{2 + 2 U_0/U_\infty}} \quad 3-50$$

Measured velocity profiles of Fekete showed that self preservation was achieved. Velocity profiles were also calculated with the viscous program for a set of conditions identical to Fekete, namely for $U_0/U_\infty = 0.95$. Analytically predicted velocity profiles also had similarity and are shown in Figure 12 to be in good agreement with measured values.

Experimental values of the excess momentum were obtained by integrating Fekete's similarity solution for the jet velocity profiles. Computed values of the excess momentum were within six percent of the measured values. The computed jet spreading rate of 0.037 agreed favorably with the measured value of 0.036, and the computed gradient of the excess momentum was within 2% of the measured values. Since the turbulence constants were not adjusted, this is very good agreement.

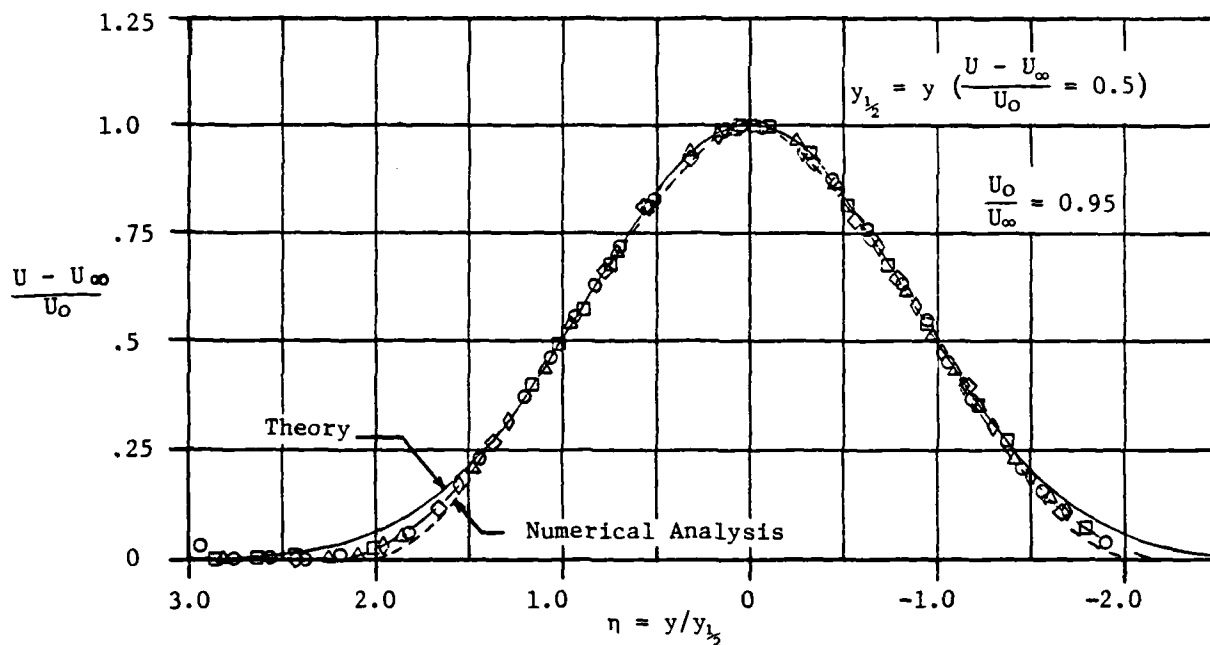


Figure 12. Comparison of Calculated and Measured Jet Velocities

MATCHING PROCEDURE

The analysis of the jet flap is matched to the analysis of the turbulent jet by iterating between solutions for the shape of the jet flap and the variation of the jet thrust, as follows: for given distributions of jet thrust and sink strength, the inviscid jet flap program yields an estimate of the jet shape. The curvature and pressure distribution along the jet sheet then provides a set of boundary conditions for calculating the jet thrust and sink strengths with the turbulent mixing program. These solutions are iterated until the computed reduction of jet thrust and the jet drag induced on the airfoil converge to within acceptable limits. The iterations are started with a classical jet flap solution in which the thrust

of the jet is assumed to be constant, and the sink strengths are all zero. Generally, three to four iterations are required to obtain convergence.

The thrust and momentum of the jet are evaluated in the following way: The net thrust on the airfoil can be calculated by integrating the computed velocity profile at any point in the wake. For simplicity, consider the case in which the jet is undeflected, as shown in Figure 13. If the upstream control surface is far enough away for the flow across it to be undisturbed, the momentum equation yields:

$$T = \int_s (P + \rho U^2) dy - \int_1 (P_\infty + \rho U_\infty^2) dy \quad 3-51$$

By factoring and recombining the terms, we get

$$T = \rho \int_{-\infty}^{+\infty} U(U - U_\infty) dy + \rho U_\infty \int_{-\infty}^{+\infty} (U - U_\infty) dy + \int_{-\infty}^{+\infty} (P - P_\infty) dy \quad 3-52$$

The continuity equation expresses the fact that the difference in mass flow between the upstream and downstream stations is equal to the mass added by the jet; that is,

$$\rho \int_{-\infty}^{+\infty} (U - U_\infty) dy = \dot{m}_j \quad 3-53$$

Substituting this relation into the momentum equation then yields the following expression for the net thrust on the airfoil:

$$T = \dot{m}_j U_\infty + \rho \int_{-\infty}^{+\infty} U(U - U_\infty) dy + \int_{-\infty}^{+\infty} (P - P_\infty) dy \quad 3-54$$

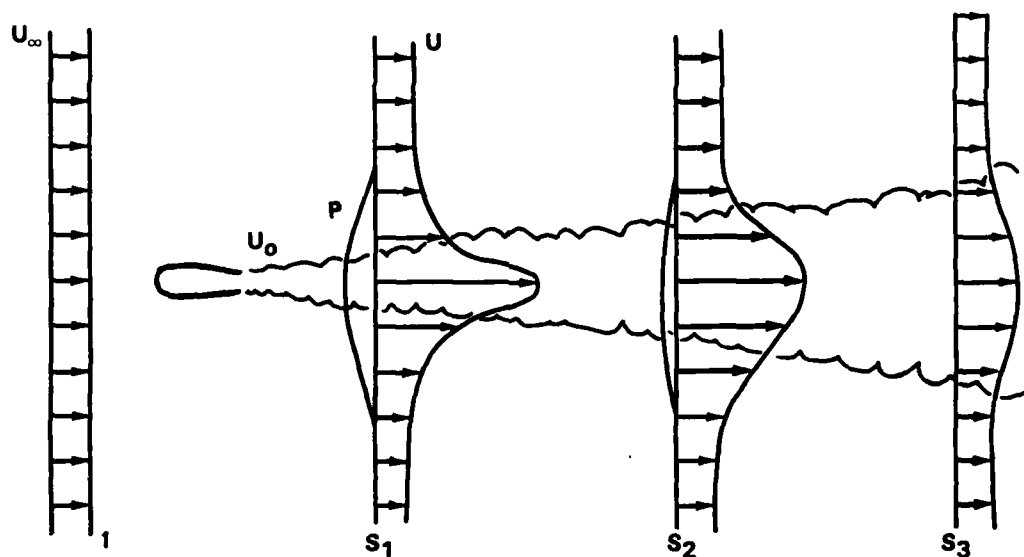


Figure 13. Control Surfaces for Evaluating Jet Thrust

The first term is the source thrust generated by the added mass of the jet. For turbojet engines, which draw the initial mass of the jet from the atmosphere (except for a very small fuel fraction) the source thrust is negligibly small. This term is also zero in a pure wake. On the other hand, for rocket engines, which generate the initial jet mass from internal fuel, this term is significant. It must also be included in evaluating the thrust of the jet flap model since the jet mass is added to the tunnel mass flow from inside the model. The second term is the integral of the excess momentum; it represents the part of the initial jet momentum which is greater than the displaced free stream momentum. In a pure wake, it would represent the drag of the body. The last term represents the pressure force due to the perturbations of the velocity field. Close to the model, this pressure term is significant. However, it vanishes at points far downstream. As it does, the excess momentum increases, because the net thrust is the same at every station behind the model.

The deflection of the jet flap is determined by the momentum flux of the jet, not the net thrust. Thus, the variation of the jet momentum flux,

$$T_M \equiv \dot{m}_j U_s + \rho \int_{-\infty}^{+\infty} U(U - U_s) dy \quad 3-55$$

is used in the inviscid program for calculating the jet shape. U_s is the velocity in the main stream on either side of the jet. An average of the values at the jet boundaries is used for the value of U_s within the jet. Outside the jet, where $U = U_s$, the integral vanishes. At downstream infinity, where $U_s = U_\infty$, the jet momentum flux does equal the net thrust of the jet.

The sink distribution represents the drag induced on the airfoil by the jet. In principle, the sinks can be determined from the streamwise variation of the excess mass within the jet (The variation of the excess mass must be used, rather than the total entrainment, because only the change in excess mass affects the flow outside the jet.) However, due to the arbitrary way in which the jet boundary is defined, it is more accurate to evaluate the sink strengths directly from the change in jet thrust.

The force on a section of the jet and the equal but opposite reaction on the airfoil are shown in Figure 14. The reaction force on the airfoil has two components: $\rho U_0 \Gamma$, a component due to the interaction between the sink and airfoil singularities, and $\rho U_\infty d\Gamma$, a component due to the change which the sink produces in the strength of the airfoil singularities. These are also shown in the figure. Since the horizontal force induced by the sink must equal the horizontal component of the thrust loss, an estimate of the sink strength at each point on the jet is obtained from the relation

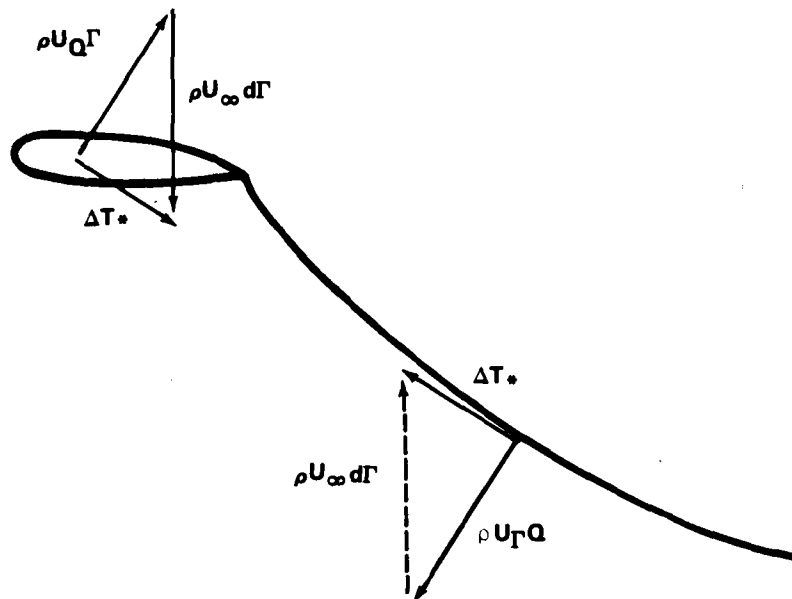


Figure 14. The Equal but Opposite Drag Forces on the Airfoil and Jet

$$\Delta T_* \cos \theta = \rho U_Q \Gamma \cos \lambda$$

3-56

in which ΔT_* is the local change in the actual thrust of the jet, and $\rho U_Q \Gamma \cos \lambda$ is the net horizontal force induced on all the other singularities by the sink. Alternately, the equal but opposite force induced on the sink by all the other singularities may be used on the right hand side. In the next inviscid calculation, the strength of the airfoil singularities are corrected to include the influence of the sink distribution. Therefore, when the iterations have converged, both components of the force on the airfoil are determined.

The local thrust of the jet, T_* , is defined to be the thrust obtained by an isentropic expansion (without additional mixing) of the local mass

of the jet to the pressure at infinity. It is necessary to use the local thrust for defining the sink strengths, rather than the local excess momentum because the excess momentum does not include the contribution of the pressure term to the jet thrust.

The local thrust of the jet is evaluated from the local jet velocities in the following way. The local thrust is defined as

$$T_{\star} = \dot{m}_j U_{\infty} + \rho \int_{-\infty}^{+\infty} U_{\star} (U_{\star} - U_{\infty}) dy \quad 3-57$$

in which U_{\star} prescribes the velocity distribution of the expanded jet. Since the expansion is assumed to occur without mixing, the continuity equation for each stream tube yields

$$\rho U_{\star} dy_{\star} = \rho U dy \quad 3-58$$

Thus, the local thrust can be written

$$T_{\star} = \dot{m}_j U_{\infty} + \rho \int_{-\infty}^{+\infty} U (U_{\star} - U_{\infty}) dy \quad 3-59$$

Then, since U_{\star} is defined by Bernoulli's equation,

$$\frac{1}{2} \rho U_{\star}^2 + P_{\infty} \equiv \frac{1}{2} \rho U^2 + P \quad 3-60$$

the local thrust can be expressed in terms of the local jet velocity distribution:

$$T_{\star} = \dot{m}_j U_{\infty} + \rho \int_{-\infty}^{+\infty} U ([U^2 + 2(P - P_{\infty})/\rho]^{\frac{1}{2}} - U_{\infty}) dy \quad 3-61$$

In the irrotational flow on each side of the jet, $U = U_s$ and

$$2(P - P_{\infty})/\rho = U_{\infty}^2 - U_s^2 \quad 3-62$$

(U_s has a different value on each side of the jet.) As a result, this integral differs from zero only within the jet so that the local thrust can be evaluated from the computed velocity profiles within the jet.

As a footnote to this discussion, Jones (Schlichting, ref 23) derived Equation 3-61 for use in determining airfoil drag from a wake survey. However, by expanding the radical in powers of $2\Delta P/\rho U^2$, it can be seen that this equation does not give the drag (or thrust) on the airfoil:

$$T_* = \dot{m}_j U_\infty + \rho \int_{-\infty}^{+\infty} U(U - U_\infty) dy + \int_{-\infty}^{+\infty} (\Delta P - \frac{1}{2} \Delta P^2 + \dots) dy \quad 3-63$$

This series converges for $\Delta P < \rho U^2$, but is equal to the actual drag only when $\Delta P \ll \rho U^2$. Thus, Jones' method does not yield the actual drag when $\Delta P \sim \rho U^2$. His error was the assumption that evaluating U_* with Bernoulli's equation gives the actual drag. To obtain the drag, U_* must be defined from the drag equation

$$\rho \int_{-\infty}^{+\infty} U_*(U_* - U_\infty) dy_* \equiv \rho \int_{-\infty}^{+\infty} U(U - U_\infty) dy + \int_{-\infty}^{+\infty} \Delta P dy \quad 3-64$$

which yields

$$U_* \equiv U(1 + \Delta P/\rho U^2) \quad 3-65$$

On the other hand, the integrand does not go to zero outside the jet with this definition of U_* , so that there is no advantage over using Equation 3-54. We have therefore used the method of Betz (Schlichting, ref 28) to measure the jet thrust in the experimental phase of this study.

Jet Drag of a Cylinder

In order to illustrate the results of this matching technique, the jet drag induced on a cylinder has been calculated. A total of 36 vortex panels were used to represent the cylinder, and 100 sink panels were used for the jet. The jet was considered to originate at the rear

stagnation point and trail straight back along the stagnation streamline. A Reynolds number of $Re = 500,000$ and jet thrust coefficient of $C_{\mu} = 1$ were used. However, it was assumed that the viscous drag of the cylinder was zero and that there was no separated wake.

A total of three iterations were required for convergence. In Figure 15 the computed surface pressure distribution for this case is compared to the computed distribution for the classical case without a jet. It can be seen that the effect of the jet is to accelerate the flow around the cylinder and reduce the base pressure. This produces a net pressure drag of magnitude $D/T_j = .207$ on the cylinder.

The streamwise variation of the excess momentum in the jet is shown in Figure 16. At downstream infinity, the excess momentum plus the

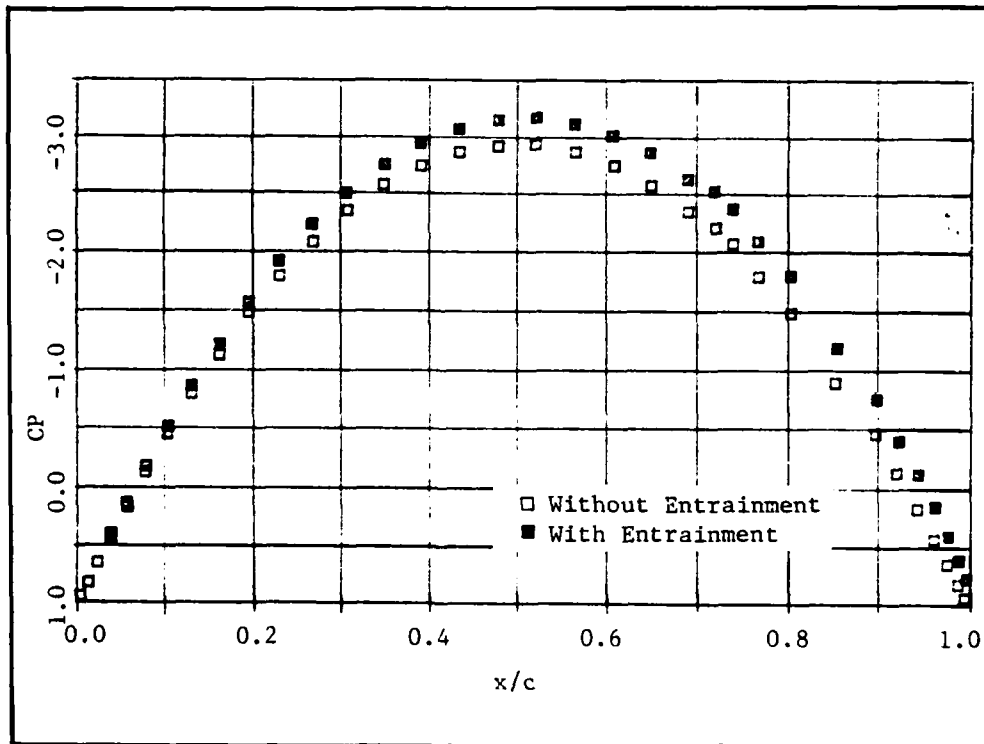


Figure 15. Comparison of Cylinder Surface Pressure Distributions

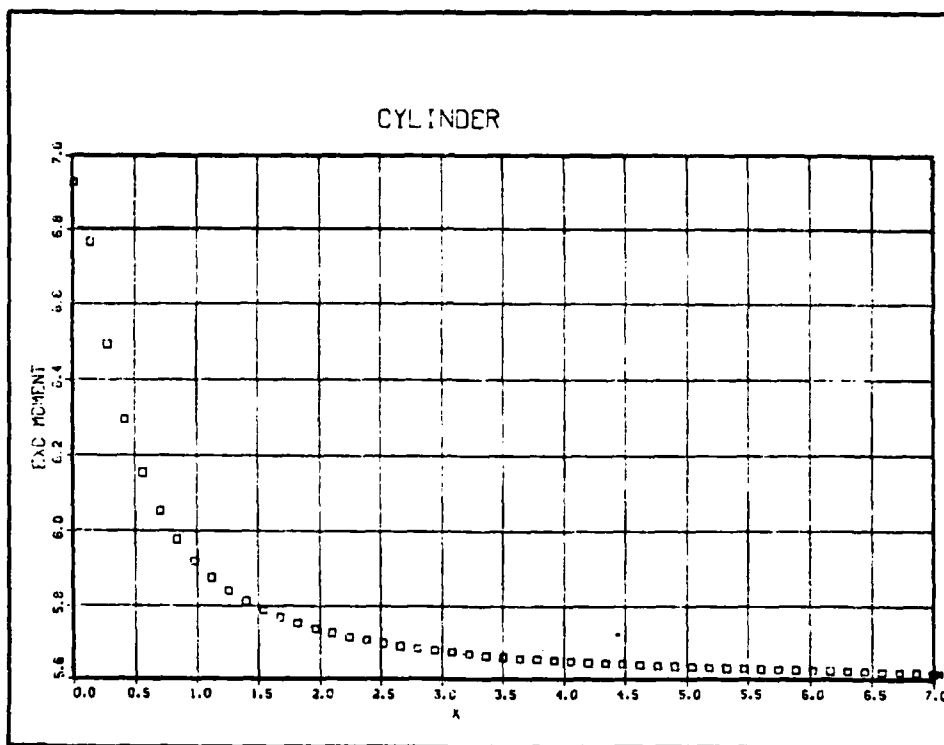


Figure 16. Variation of Excess Momentum in the Jet

source thrust give the net force on the cylinder. By $X/D = 7$, the reduction in jet thrust is $\Delta T/T_j = .189$. Considering the difficulty in accurately determining the drag from surface pressure integrations, this was felt to be good agreement, and additional iterations were not performed. A solution was also attempted using the sink strengths given by the total jet entrainment, but the jet drag on the cylinder approached a value near $D/T_j = 0.5$, which was roughly twice the change in jet thrust.

IV. EXPERIMENTAL METHODS

WIND TUNNEL FACILITY

Tunnel Description

The wind tunnel in which the jet flap model was tested is a two-dimensional tunnel designed especially for powered lift testing. A photograph of this tunnel is shown in Figure 17. The tunnel is an open circuit return type driven by a downstream multi-tube ejector.

The test section is 51 cm wide and 366 cm high. With the 20 cm chord model of this test, the height-to-chord ratio for this tunnel is 18.0. This ratio minimizes wall effects and allows sufficient space for the jet to develop downstream for the large jet deflections and momentum coefficients tested. For example, with $C_{\mu} \sim 50^\circ$, the velocity overspeed is estimated to be $1.001 U_\infty$. The test section extends approximately 8 chord lengths (158 cm) upstream and 12 chord lengths (23 cm) downstream of the model quarter chord. A smooth entrance to the test section is provided by a 61 cm long elliptical bellmouth. The vertical walls of the tunnel diverge from 51 cm apart at the downstream end of the bellmouth to 55 cm apart at the tunnel exit to allow for wall boundary layer growth. The model was mounted in the test section 213 cm above the tunnel floor. The tunnel was constructed of plywood and lumber.

A multi-tube ejector was located at the downstream end of the test section to power the tunnel. The ejector primary nozzle arrangement consisted of 304 separate .457 cm I.D. nozzles uniformly spaced in the ejector plane. These nozzles were fed by two 10 cm diameter steel plenums, located outside the tunnel, one on either side. The ejector gives an entrained to primary mass flow ratio of 20. With the air



Figure 17. Two-Dimensional Wind Tunnel

supply available for this test, the maximum tunnel speed was approximately 24 meters per second.

The model was mounted in aluminum plates set flush in the tunnel's wooden walls. These plates contained circular insets which allow the model to be rotated to angles of attack between -10 and $+20$ degrees. Windows were provided in the insets to review the model during testing. A photograph of the aluminum plates is shown in Figure 18.

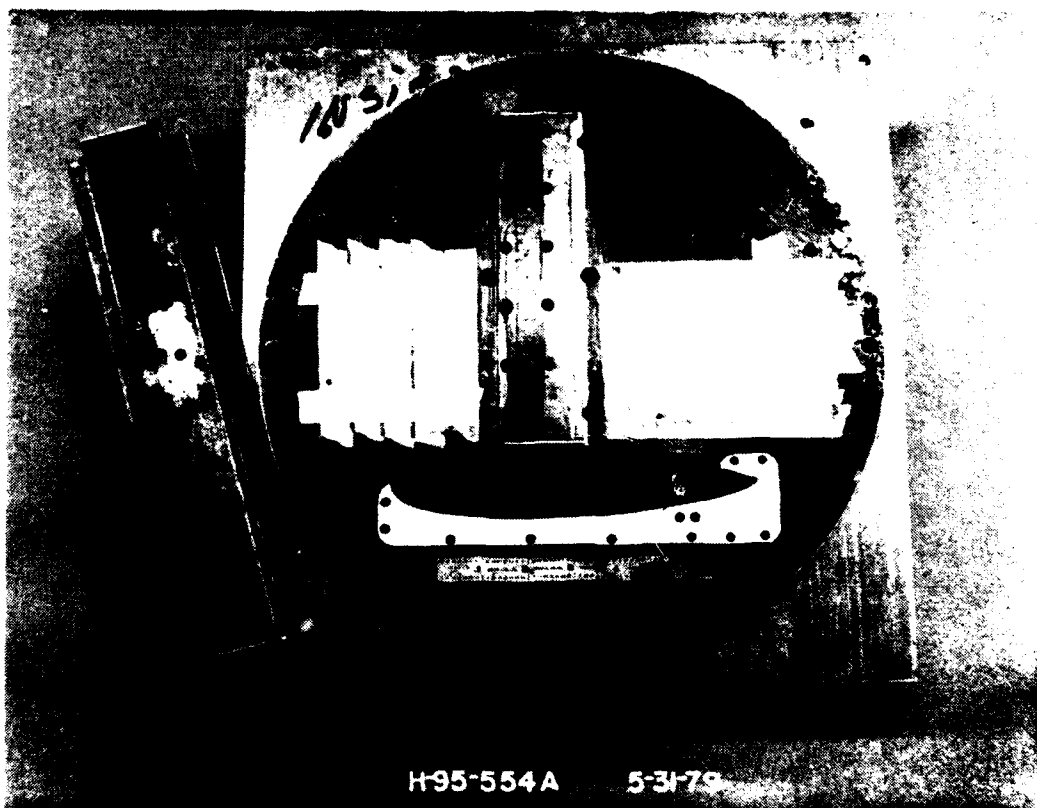


Figure 18. Airfoil Model Endplates

Two boundary layer blowers were mounted on each test section side-wall. A 30 cm long BLC blower was located 7.6 cm upstream of the model leading edge and was set at an angle of 15 degrees from the vertical. A 15 cm long blower was attached to the model mount inserts, with the nozzle located at the midchord blowing parallel to the chordline. All four blowers were fed from a common supply line and operated at the same supply pressure. The blower nozzle gaps were set to .100 cm.

Tunnel Calibration

Calibration of the wind tunnel test section was performed in two steps. The first stage was to adjust the vertical wall divergence to obtain a constant static pressure distribution along the entire length of the test section. Then the total and static pressure distributions in the area where the model would be located were surveyed. This survey data was used to generate a calibration curve of test section conditions versus a measured sidewall surface static pressure. The results of the calibration procedures are discussed below.

A 5 cm diameter pipe containing static pressure taps every 15.25 cm was mounted equidistant between the tunnel walls and 213 cm above the tunnel floor (at the future model chord line). The first static tap was at the downstream end of the bellmouth at the inlet. The tunnel walls were set with a total of 3.8 cm divergence in width (51 cm at the entrance, 55 cm at the exit). Longitudinal static pressure distributions were measured for a range of tunnel speeds. These showed no systematic variations.

The total and static pressure distributions in the region of the model locations were measured in the second phase of the tunnel calibration. A rake consisting of 10 total pressure tubes mounted 5 cm apart and 6 static pressure tubes mounted 5 cm above the total pressure tubes was used. The rake data were taken from 30 cm above the model location to 40 cm below the model location at 5 cm intervals. The resulting pressure data are shown in Figure 19.

The tunnel sidewall static pressure was measured at taps on each sidewall approximately 10 cm downstream of the bellmouth and 2.3 cm above the tunnel floor (on the model chord line). This sidewall static pressure was measured at each tunnel set condition at which the test section pressure distributions were taken. These data were used to generate the tunnel speed calibration factor which relates the tunnel dynamic pressure to the sidewall static pressure drop below atmospheric pressure. This calibration

$$q = 1.03 (P_q - P_s)$$

4-1

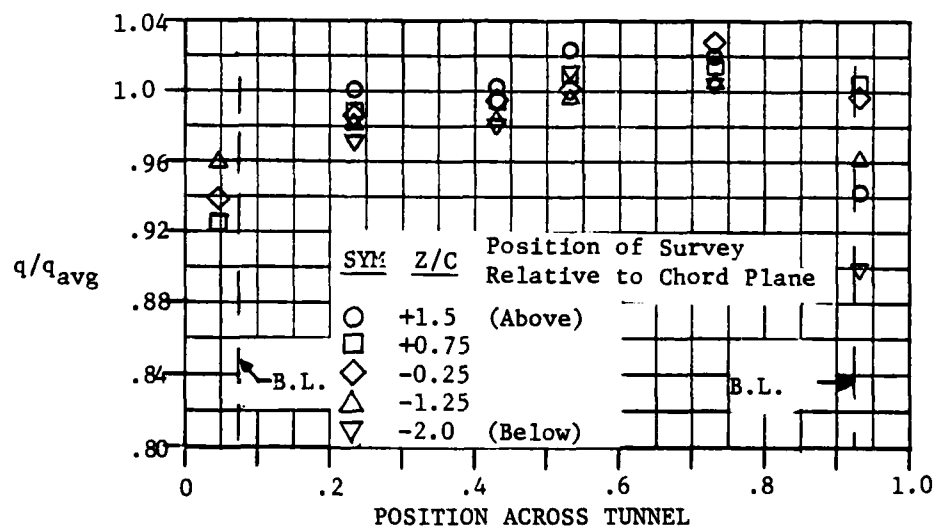


Figure 19. Dynamic Pressure Distribution At Model Location

AIRFOIL MODEL

Model Description

The airfoil model had a thickness to chord ratio of 0.20 and was symmetrical except in the area of the blowing slot and upper flap contour. The model chord was 20 cm and the installed model span was 51 cm. A sketch of the model cross section is shown in Figure 20 and a photograph is shown in Figure 21.

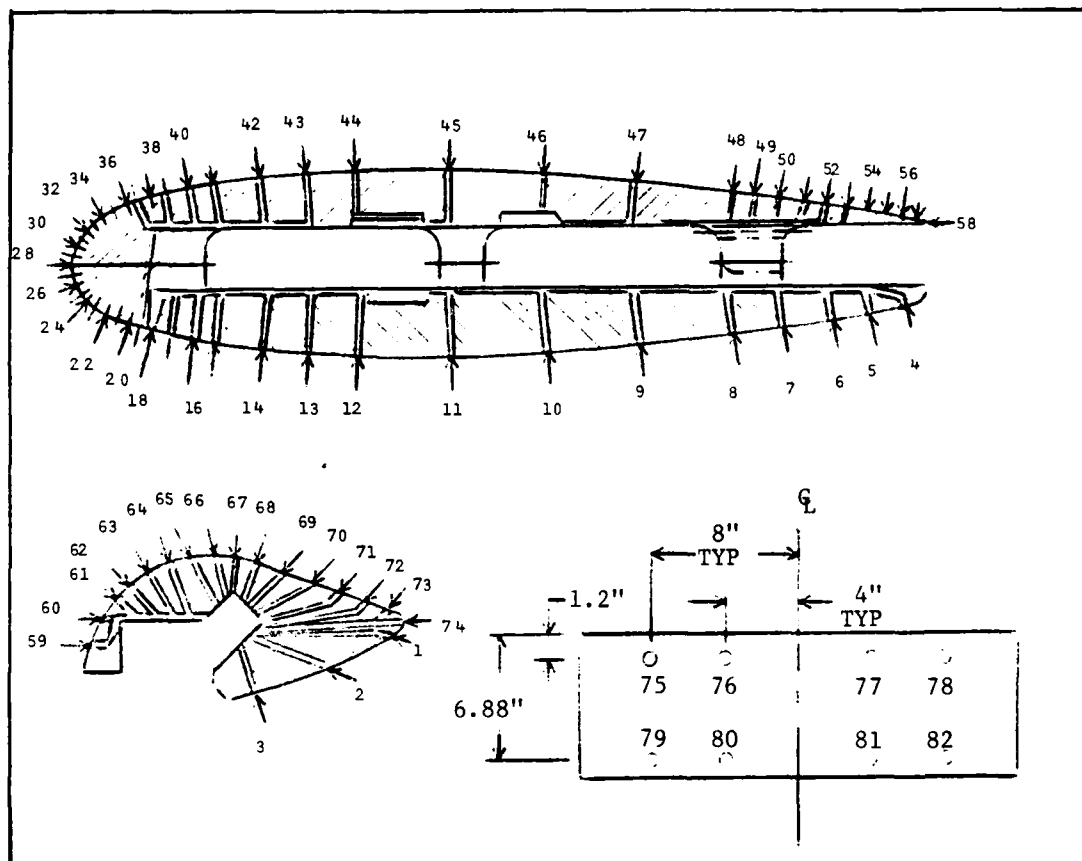


Figure 20. Location of Pressure Taps on Airfoil Model



Figure 21. Airfoil Model Mounted on Wind Tunnel Balance

Forward of the maximum thickness ordinate ($X/C = 0.4$), the model contour is described by an ellipse which gave an effective leading edge radius of approximately 5.5% of the chord. Aft of the maximum thickness, a different ellipse was used to describe the contour. The jet blowing slot was located on the upper surface at $X/C = 0.9$. The upper nozzle lip had a thickness equal 0.005% of the model chord. The 10% chord flap had a circular Coanda turning surface with a radius equal to 1.4 cm which was tangent to the flat upper surface trailing edge at $X/C = 0.93$. The airfoil trailing edge was slightly blunt, with a thickness equal 0.01% of the airfoil chord. The lower surface trailing edge was modified slightly from the ellipse to give a sharp edge at the trailing edge. The model was constructed of aluminum. Model contours in the center 5 cm of span were held to ± 0.005 cm.

The model flap was attached to the forward portion of the model with one of five separate flap brackets. These brackets were attached internally, and did not break the airfoil mold line. The flap could be mounted at angles of 0° , 10° , 30° , 50° , and 70° relative to the airfoil chord plane. The flap upper surface made an angle of 20° relative to the chord. Thus, the flap surface and hence the nominal jet deflection angle could be set to 20° , 30° , 50° , 70° , and 90° relative to the airfoil chord. The effective flap pivot point is at $X/C = .9$ $Y/C = 0.05$.

The total span of the constant model cross section was 61 cm. The model therefore pierced the tunnel walls and the nozzle gap was plugged to the correct tunnel span of 51 cm. The nozzle plugs installed at each end were given a bellmouth shape in the planform to maintain a constant nozzle total pressure distribution up to the wind tunnel walls.

A large portion of the model interior was open and acted as the nozzle feed plenum. The flow passage from the plenum to the nozzle convergence was designed with two 90° turns with flow areas approximately twice the nozzle area. This was done to improve the spanwise jet total

pressure distribution. The convergence to the nozzle exit was formed by a flat upper surface and the circular extension of the Coanda surface. The flap brackets were designed such that the nozzle convergence was the same for all flap settings. The nozzle exit was located at 90% chord and designed to have the jet exit parallel to the airfoil chord. Ambient temperature compressed air was supplied to the model through steel endplates to which were welded 10 cm pipe fittings. A portion of the right steel endplate could be removed to ease flap changes. Sealing of all mating surfaces was provided by O-rings.

The model was instrumented with a total of 82 pressure taps. Seventy-four taps were arranged in a streamwise row on the airfoil and flap at mid-span. Of these, two taps were located on the blunt, aft facing surfaces, the nozzle lip, and the flap trailing edge. An additional eight pressure taps were arranged in two spanwise rows to monitor test two-dimensionality. Figure 20 shows the location of all of the pressure taps.

Model Calibration

The model's jet nozzle pressure distribution and actual jet reaction force and angle were calibrated prior to installation in the two-dimensional tunnel.

The nozzle was plugged to the correct span as discussed in the model description section. The model was then mounted on the six component pyramidal balance located in the low speed wind tunnel at Rockwell International's Columbus Plant. The air supply plumbing to the model closely matched that to be used in the 2-D tunnel installation. A venturi and associated instrumentation were installed in the air supply line. A pressure tap in one of the steel model end fittings was used as the calibration reference pressure. One objective of the calibration was to generate a curve relating this reference pressure to the nozzle exit total pressure.

Nozzle exit total pressure distributions were measured at each flap angle for a range of pressure ratios from 1.1 to 2.2. These distributions were quite uniform. At each test condition, the model plenum reference pressure was recorded. These data were then used throughout the test to compute exit total pressure.

For each flap angle, a set of force data were taken over a pressure ratio range of 1.1 to 2.1. The resultant thrust angle was computed from the model lift and drag forces and this is shown in Figure 22. Model isentropic thrust was calculated from the venturi measured mass flow and the nozzle isentropic velocity. The isentropic velocity was computed assuming an expansion from the nozzle total pressure (obtained from the calibration curve) to the ambient pressure. Jet temperature was measured at the venturi as experience has shown this to be within a few degrees

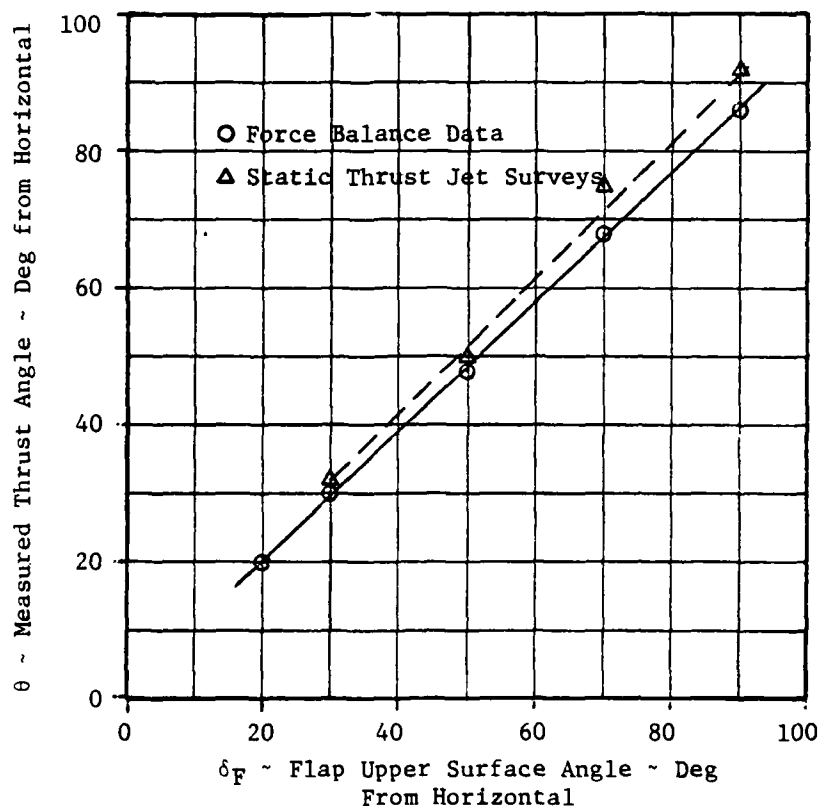


Figure 22. Comparison of Balance and Survey Measured Thrust Angles

Rankine of the temperature at the nozzle exit. Figure 23 shows the variation of the ratio of the actual thrust to the isentropic thrust as a function of the model flap angle.

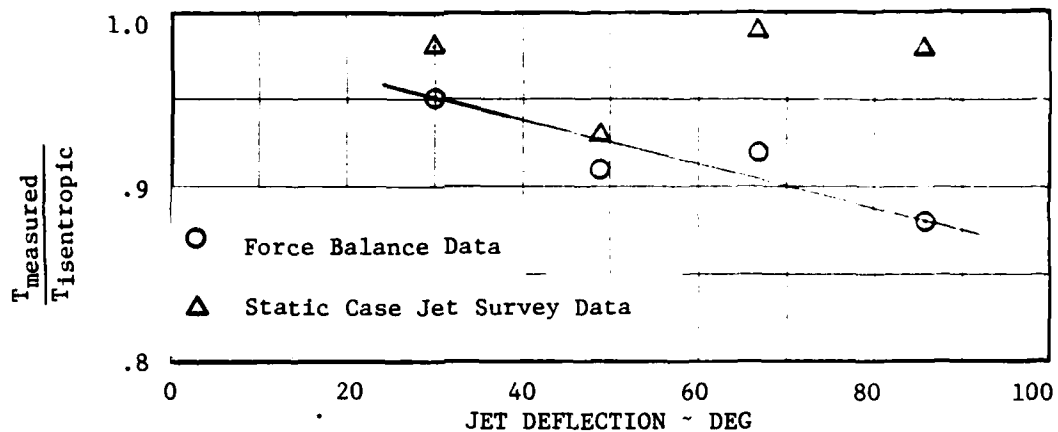


Figure 23. Jet Flap Nozzle Thrust Calibration

INSTRUMENTATION AND EQUIPMENT

Survey Probe

Downstream jet/wake surveys were made using a United Sensor three-dimensional directional flow probe, type DC-125. This probe has a stepped conical sensing head with one pressure tap at its center and four taps located on the conical surface. This probe was calibrated as described below to accurately measure total and static pressure and flow angle over a flow angle range of ± 50 degrees in pitch and ± 20 degrees in yaw. A photograph of this probe is shown in Figure 24.



Figure 24. Five Port Directional Flow Probe

The probe was calibrated by testing it in a jet of air of known total and static pressure and flow direction. Data were taken at a matrix of probe angles of ± 50 degrees in pitch and ± 20 degrees in yaw, in 5° increments. At each position, the five separate probe pressures were recorded. These data were repeated in calibration jets of 2.6, 5.2, and 7.75 cm Hg gage total pressure. At each point, the following calibration constants were calculated:

$$\begin{aligned} KP &= \frac{P5-P1}{P5-P3} - 1 \quad (+ \text{ Pitch, i.e., } P1 < P3) \\ &= \frac{P3-P5}{P5-P1} + 1 \quad (- \text{ Pitch, i.e., } P1 > P3) \end{aligned}$$

$$\begin{aligned} KY &= \frac{P5-P4}{P5-P2} - 1 \quad (+ \text{ Yaw, i.e., } P4 > P2) \\ &= \frac{P2-P5}{P5-P4} + 1 \quad (- \text{ Yaw, i.e., } P4 < P2) \end{aligned}$$

$$KPT = \frac{P5-PT}{PT-PS} \quad \begin{array}{l} (PT = \text{jet total pressure} \\ PS = \text{jet static pressure}) \end{array}$$

$$KPS = \frac{P5 - \frac{P1+P2+P3+P4}{4}}{PT-PS}$$

These data are then used to generate calibration matrices for pitch angle (function of KP and KY), yaw angle (function of KP and KY), total pressure factor (KPT as function of pitch and yaw angle), and static pressure factor (KPS as function of pitch and yaw angle).

In an unknown flow field, the five probe pressures are measured, and used to compute KP and KY. These values are used as the independent variables in two dimensional interpolations in the calibration matrices to give the pitch and yaw angles of the flow. These two angles are then used as the independent variables in two dimensional interpolations to give the flow's total and static pressure factors, KPT and KPS. These factors and the five probe pressures are used to compute the unknown flow's total and static pressure.

The four required calibration matrices were generated using the data taken in the 7.75 cm Hg total pressure jet. The calibration was then checked by using the calibration to reduce the data taken at the other two jet pressures. Flow angles were generally good to ± 2 degrees. Total static pressures were within $\pm .05$ cm Hg.

During the test, the 3-D probe was mounted on a traversing rig attached to the tunnel sidewall. This rig consisted of two support rails and a motorized screw jack which could position the probe support arm anywhere along 76 cm travel. The motor was under either manual or computer control as discussed in the test procedure section. The traversing rig could be bolted to the tunnel wall at any location. The probe could be positioned at any spanwise location and at eight discrete downstream locations for any rig position.

Instrumentation

The following data channels were used during the 2-D wind tunnel test:

- Wind tunnel sidewall static pressure
- Model plenum calibration pressure
- Model air supply venturi upstream pressure
- Model air supply venturi differential pressure
- Model air supply venturi temperature
- Model surface pressures - 5 module scanivalve
- Jet/wake 3-D survey probe - 5 low range and 5 high range transducers
- Jet/wake 3-D survey probe traversing rig position
- Two-dimensional wind tunnel ejector supply pressure
- Test section sidewall boundary layer control blower plenum pressure

All transducers were calibrated prior to the first test series and were recalibrated prior to the second test series.

All instrumentation output signals were amplified and filtered before going to the multiplexed analog to digital converter in the IBM 1800 data acquisition computer. Readings were then converted to engineering units and preliminary integrations of lift (from surface pressures) and drag (from both wake data and surface pressures) were performed.

The model surface pressures were assigned to the various Scanivalve modules on the basis of expected maximum pressures. Each Scanivalve module had the transducer reference pressure (the tunnel sidewall pressure) connected to the first and last ports. Each model pressure connected to the two low range (± 1 cm Hg max. input) modules was connected to two consecutive scanivalve ports. This was done to give an effective settling time of twice the normal settling time which was two seconds.

The data acquisition program was written to read every other port on these two modules. The maximum number of ports read was set by the higher range modules which then allowed this technique to be used.

The 3-D wake probe was required to measure the flow properties in both the low total pressure tunnel airflow and the higher total pressure jet sheet. To improve the accuracy of the readings in the low pressure flow, the following scheme was used. Each of the five probe pressures were routed to two transducers, one set for approximately .775 cm Hg maximum input (data system saturation limit) and the other set for approximately 26 cm Hg max input. A pressure relief valve was installed in the tube leading to the low pressure transducer. This valve was set to shut off this pressure line at about 1.5 cm Hg, well below the 2.5 cm Hg transducer physical limit. With this setup, in the low pressure tunnel flow, each pressure was read on two transducers, and the data reduction program used the output from the lower range transducer. As a probe entered the jet, the low range transducer data channel reached saturation, but the relief valve protected the transducer from over pressure damage. The data reduction program recognized this saturation value and instead used

the high range transducer output for calculations. As the probe emerged from the jet, the relief valve reopened, the low range transducer fell below saturation, and its more accurate output was again used for that pressure. The over-lapping ranges of transducer electrical saturation, relief valve operation, and transducer physical pressure limits allowed the scheme to work well in the region of large pressure fluctuations at the edge of the jet.

TEST PROCEDURES

Data Acquisition

The major problem with two-dimensional testing is the effect of the sidewall boundary layer. The sectional wing lift is reduced in the wall boundary layer which gives rise to three-dimensionality and the associated induced drag. A more serious effect may occur when the wall boundary layer is forced to separate by the adverse pressure gradients on the model. The strength of these adverse gradients is significantly increased by the large lifts generated by a jet flap device.

There are several ways to mount a two-dimensional model on a balance, but each has significant problems for this particular test. If the entire model is metric (on the balance), with or without a portion of the sidewalls, the measured loads will include the non-uniform load near the walls due to the boundary layer. In this case, a decision must also be made whether or not to attach a portion of the walls to the model. In the first case, the tip sealing problem is alleviated, but the forces on the endplates must be accounted for. In the second case, a force free seal must be made at the model tip/wall juncture.

A second approach is to mount only a center span portion of the model on the balance. This eliminates the direct force effects of the tip loading, but again introduces joints and seals, near the region of interest. This approach is substantially complicated on a jet flap model by the need to segment the nozzle.

Any balance installation on a jet powered model faces the problem of bridging the balance with the air supply pipes. This problem has been successfully solved many times in the past. But each installation is generally unique and there is usually a development phase before a

useable set-up is obtained. In any case, the balance must be calibrated with the air supply system installed and pressure effects must be considered. All these effects reduce the two-dimensionality of the pressure distributions and measured forces.

In view of the above complications, the decision was made to obtain lift from airfoil surface pressures and drag from wake surveys. Integration of the drag component of the surface pressures was also performed as a secondary drag calculation. Use of this technique also provides information on the qualitative nature of the flow.

Each run during these tests acquired data for one test condition (angle of attack, jet deflection, and momentum coefficient). Model angle of attack was set by rotating the circular wall inserts. The wake probe traversing rig was positioned such that the probe would pass the desired distance downstream of the model (approximately 15 cm). The rig tilt angle was set such that the expected flow angles would be within the probe's acceptance limits (± 50 degrees from the probe axis). The desired tunnel speed was set by adjusting a pressure regulating valve while monitoring the sidewall static pressure displayed on a digital voltmeter (DVM). Model thrust was set by operating a pressure control valve to achieve the precomputed model plenum reference pressure also displayed on a DVM. Sidewall BLC operating pressure was set in the same way as the model pressure. Proper set pressures were determined by observing tufts attached on the sidewall and model upper surface and by monitoring the model spanwise row of pressure taps.

With the desired test conditions set, data acquisition was under computer control. Model and tunnel operating conditions were read followed by the scanivalve data. The computer then commanded the traversing rig travel based on a predetermined number of datapoints and desired distance between points. Different numbers of points and point spacing could be requested in each of three survey regions (nominally above the jet, in the jet, and below the jet). After each step of probe travel, a pressure setting time of two seconds was allowed before data were acquired.

For each channel of data acquired, the computer took ten readings over a 1/2 second period. These readings were averaged and this average was stored for later conversion to engineering units.

Each run required approximately 15 minutes and required about 190 wake probe positions. Test conditions were held constant during this time by two technicians. The test engineer was able to monitor these conditions and progress of the test from the data acquisition system.

Two test series were run several months apart. The objective of the first series was to obtain lift and drag data for jet deflection angles of 30°, 50°, 70°, 90°, and momentum coefficients between 0.5 and 24.0. The objective of the second test was to survey the jet at five downstream locations for a range of test conditions to quantify the jet thrust and mass flow changes as the jet develops downstream.

Data Reduction Procedures

The following section discusses the data reduction procedures for the wind tunnel and model operating conditions, airfoil surface pressure lift and drag integrations, wake drag integration, and jet profile thrust and mass flow integrations.

The wind tunnel operating conditions are computed from the measured sidewall static pressure and ambient pressure and temperature. Because of the suck down type tunnel design, the test section total pressure is equal to the ambient pressure; this was confirmed by the test section calibration.

$$P_{T_{\text{Test Section}}} = P_{\text{ambient}}$$

This calibration provided the following relationship between the sidewall static pressure and the calibrated test section uncorrected dynamic pressure.

$$q_{\text{test section uncorrected}} = 1.03 \times (P_{\text{sidewall}} - P_{\text{ambient}})$$

A solid blockage correction was included to give the corrected test section dynamic pressure.

$$q_{\text{test section}} = q_{\text{uncorrected}} (1.0 + .0006)^2$$

Test section static pressure is then:

$$P_{\text{test section}} = P_{T_{TS}} - q_{TS}$$

Test section Mach number

$$M_{TS} = \sqrt{\left[\left(\frac{P_T}{P} \right)^{.28571} - 1 \right] \times 5} \quad 4-2$$

Model Reynold's number (based on model chord):

$$R_n = 10^6 (1.798) q_{TS} \left[\frac{1+.2M^2}{M} \right] \left[\frac{T_{TS} + 1.986 (1+.2M^2)}{T_{TS}^2} \right] \quad 4-3$$

Where T_{TS} is ambient temperature.

The model jet isentropic thrust was computed from venturi measured mass flow and jet isentropic velocity. The venturi data (P_{upstream} , $P_{\text{differential}}$, and Temperature) were reduced to a mass flow using standard computations. Isentropic velocity was computed as follows:

The nozzle exit total pressure was obtained from the model plenum calibration pressure and the calibration. The jet temperature was assumed to be equal to the measured venturi temperature as discussed earlier. The isentropic velocity is then calculated assuming isentropic expansion from the nozzle total pressure to the test section static pressure.

$$V_{\text{isen}} = \sqrt{\gamma \cdot R \cdot T_v} \sqrt{5 \cdot \left[1 - \frac{P_{\text{Test Section}}}{P_{T_{\text{nozzle}}}} \right]^{.28571}} \quad 4-4$$

Where γ is the ratio of specific heats and R is the universal gas constant.

The model jet Mach number was computed:

$$M_j = \sqrt{5 \times \left[\left(\frac{P_{T_{\text{nozzle}}}}{P_{\text{Test Section}}} \right)^{\frac{28571}{\gamma}} - 1 \right]} \quad 4-5$$

($M_j \leq 1.0$)

and used in the calculation of a theoretical mass flow:

$$\dot{m} = \sqrt{\frac{\gamma}{RT_v}} \left[\frac{A_{\text{nozzle}} \cdot P_{T_{\text{nozzle}}} \cdot M_j}{(1.0 + .2M_j^2)^3} \right] \quad 4-6$$

Comparing the theoretical and venturi measured mass flows gave a further check on instrumentation and test setup.

The model momentum coefficient is then defined:

$$C_\mu = \frac{M_{\text{venturi}} \cdot V_{\text{isen}}}{q_{TS} \cdot S_{\text{ref}}} \quad 4-7$$

The pressure coefficients were then integrated as functions of their X/C and Y/C locations to give lift and drag force coefficients, respectively.

Each integration was performed piecewise over surface of the airfoil.

A complete set of plotted airfoil pressure coefficient data from the first test series (Test 272) is included in Appendix I.

Each of the wake/jet sheet surveys from both test series has been integrated to give an airfoil drag coefficient. This integration was performed using the method of Betz, (Schlichting, ref 28), modified to include the effect of the large flow deflection angle. Direct application of the Betz method of wake survey integration does not give the correct value of drag, if the survey is made before the jet has completely turned

to the free stream direction. In that case, there is a static pressure difference which makes a contribution to the horizontal force. This contribution must be equal to the difference between the horizontal component of the local jet thrust and the thrust of the completely deflected jet. Since the jet approaches the free stream direction asymptotically, the actual thrust can, in principle, only be determined from a jet survey at downstream infinity, or from a survey which extends to infinity above and below the jet. However, if the jet is nearly parallel to the free stream, the tangential component of the jet thrust is equal to the value recovered at infinity, within experimental error. This will be seen in the analytic results. Thus, the measured tangential thrust was used to determine the thrust recovery.

V. RESULTS AND DISCUSSION

SURFACE PRESSURE DISTRIBUTIONS

The character of the measured surface pressure distributions changes as the jet thrust coefficient and deflection angle are increased. For low thrust coefficients ($C_\mu < 1$) and small deflection angles ($\theta = 30^\circ$), the pressure distributions have a saddle shape, typical of that predicted by classical, inviscid jet flap theory. A representative case is shown in Figure 25. The saddle shape is due to the suction peaks at the leading

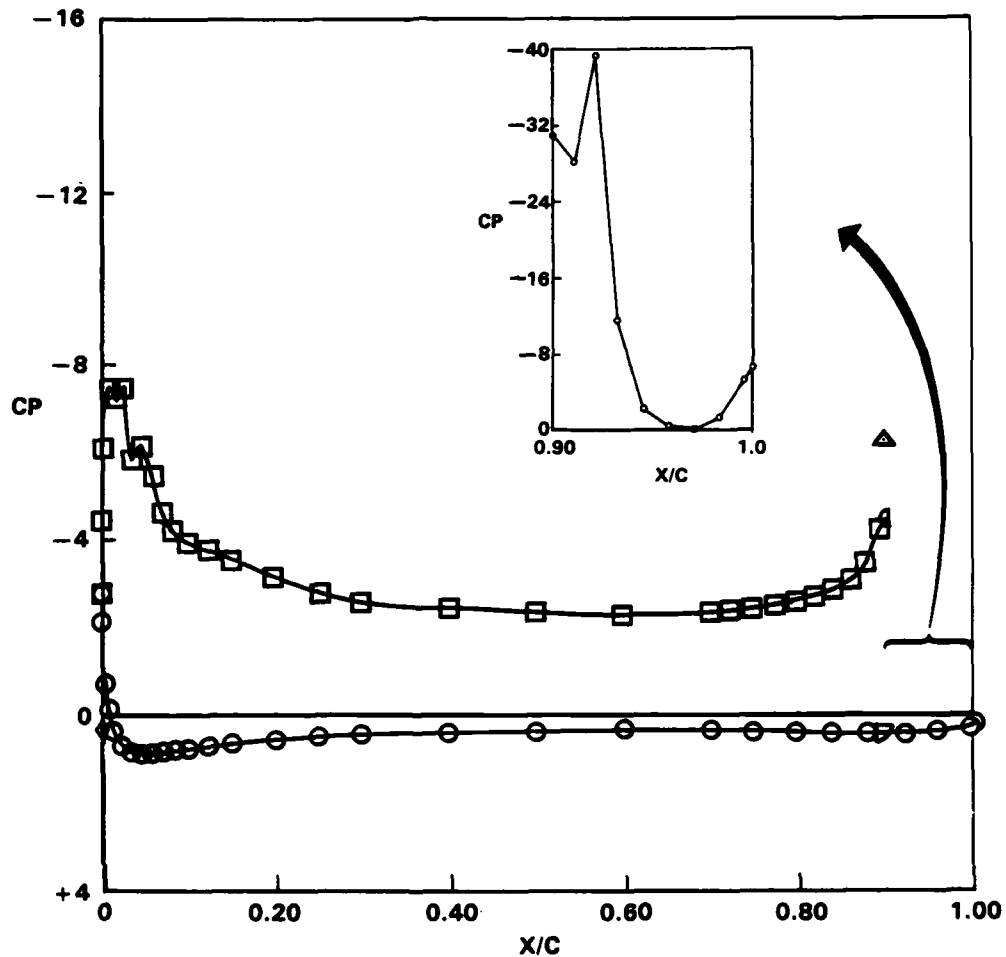


Figure 25. Surface Pressure Distribution for Attached Flow, $C_\mu = 2$, $\theta_0 = 30^\circ$

edge of the airfoil and the knee of the flap. The pressure is further reduced on the upper surface of the flap by the suction which deflects the jet over the Coanda surface. Since the radial pressure gradient balances the inertia of the jet, the reduction in flap surface pressure is given by

$$\Delta C_p = C_{\mu} c / R \quad 5-1$$

The sudden pressure rise at the point where the surface of the flap changes from a circular arc to a straight section causes a separation bubble to develop; the suction peak near the nozzle exit is due to this bubble. The smaller rise at the trailing edge of the flap is due to the jet flap effect: The jet leaves the airfoil along a tangent to the upper surface of the flap, and therefore deflects the trailing edge streamline relative to the mean line of the airfoil. This produces a suction peak similar to the one which occurs at the knee of a flap.

For moderate values of the thrust coefficient ($2 < C_{\mu} < 8$) and deflection angles ($\theta = 30^\circ, 50^\circ$), a separation bubble appears on the upper surface of the airfoil. The flow separates from the leading edge of the airfoil and reattaches at some point on the upper surface. In this case, the pressure is nearly constant over the front of the airfoil, but returns to the ideal saddle shape over the rear section. A typical example is shown in Figure 26.

For large values of the thrust coefficient ($C_{\mu} > 8$) and large deflection angles ($\theta > 50^\circ$) the separation bubble does not reattach to the surface of the airfoil. In this case, the pressure distribution is flat or even slightly arched, as seen in Figure 27. Since the combined effect of the jet and flap accelerates the flow over the unblown section at the rear of the airfoil, trailing edge separation did not occur for any case tested. The complete set of surface pressure distributions may be found in Appendix I.

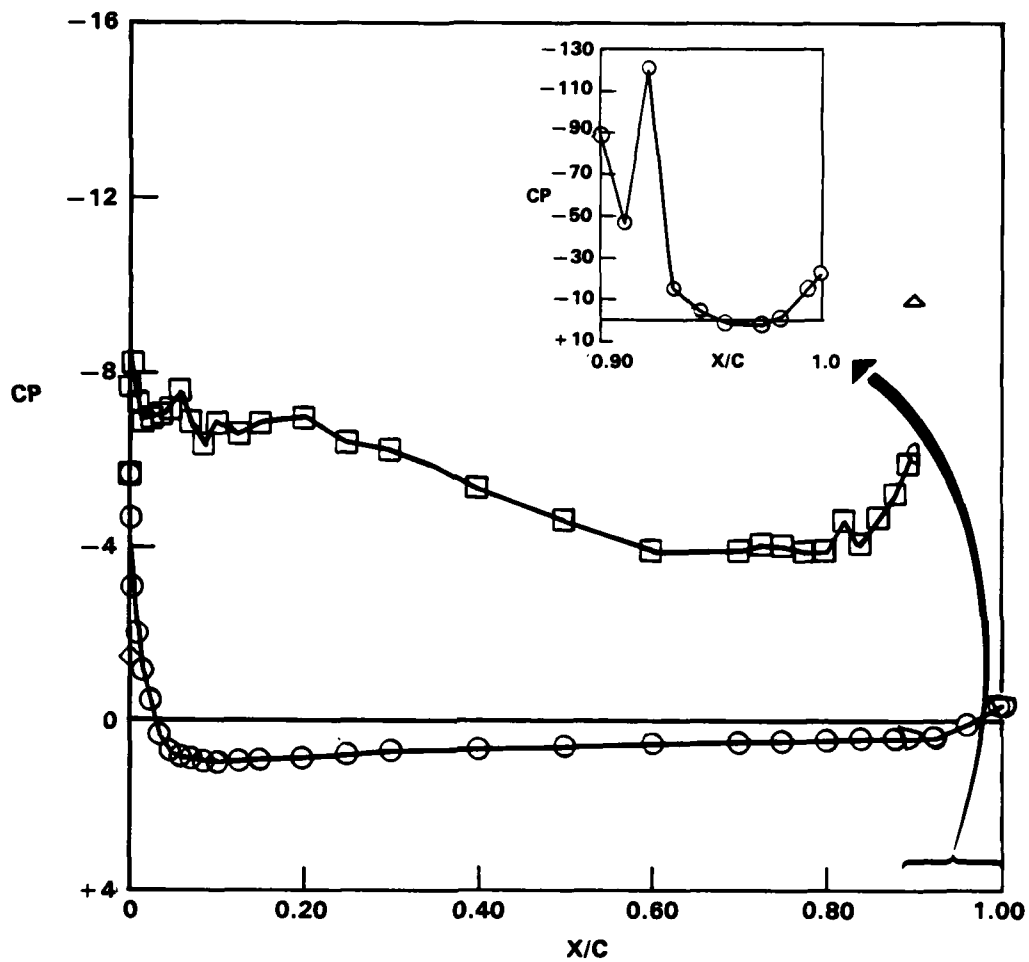


Figure 26. Surface Pressure Distribution for Leading Edge Separation, $C_{\mu} = 6$, $\theta_o = 30^\circ$

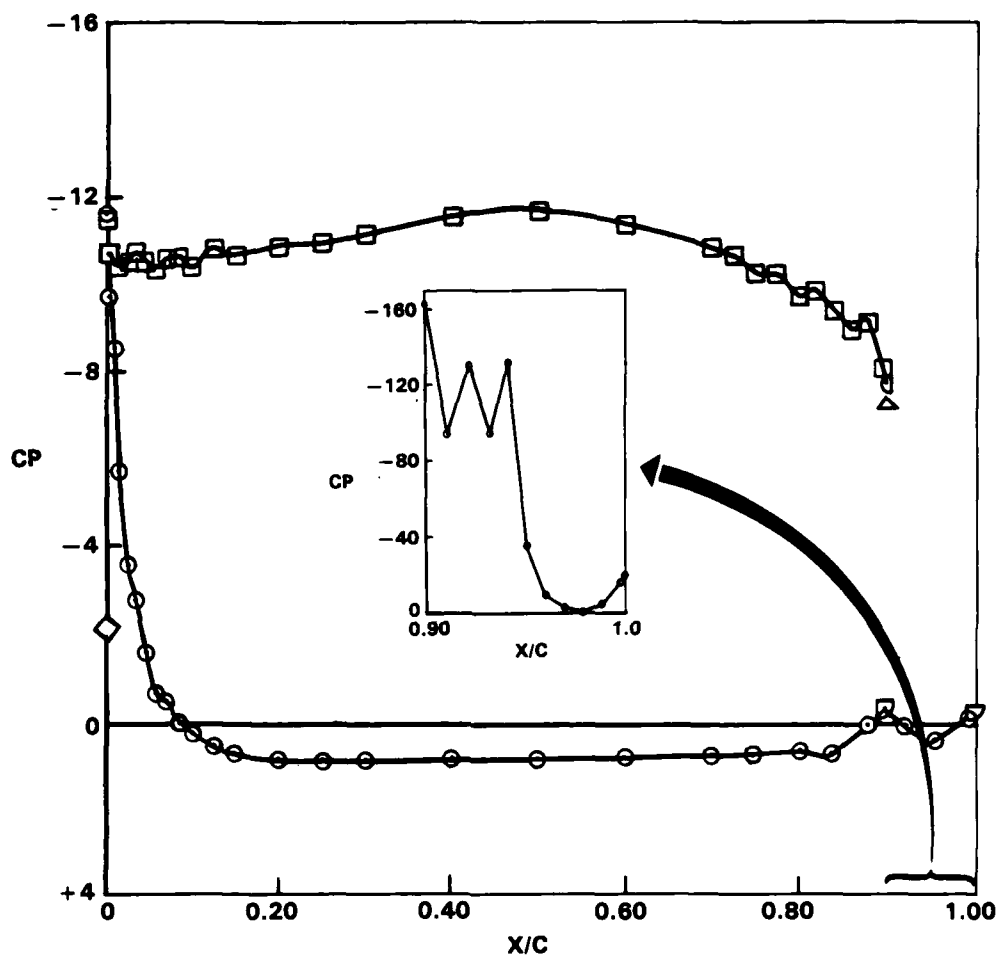


Figure 27. Surface Pressure Distribution for Upper Surface Separation, $C_{\mu} = 8$, $\theta_o = 50^\circ$

WAKE PRESSURE DISTRIBUTIONS

Distributions of the total and static pressure, as well as the flow angularity, were obtained in the near wake. The trend of the wake data was consistent with the surface pressure distributions. A representative set of pressure profiles for the case in which the flow did not separate from the airfoil is shown in Figure 28. The total pressure profile has the classical bell shape, and the static pressure is less above the jet than below it. In Spence's (ref 10) theory, this pressure jump is represented by the vortex sheet. The reduction in static pressure within the jet is due to the jet turbulence, which causes an actual reduction

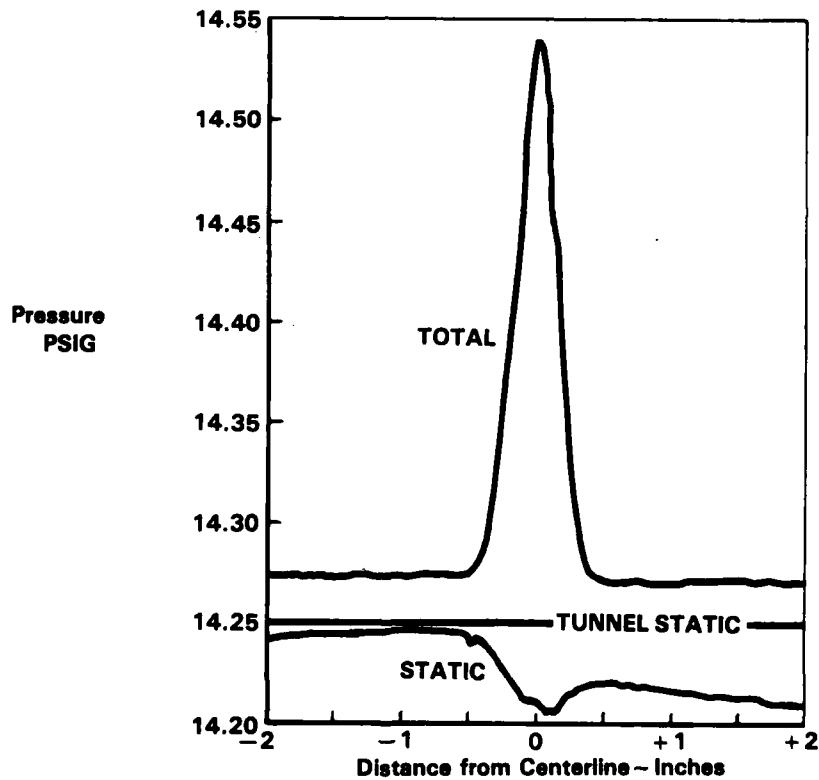


Figure 28. Total and Static Pressure Surveys in the Jet Wake for $C_u = 1$ and $\theta = 50^\circ$

in the pressure and introduces an error in the probe reading. Since the error is small, no correction was used. When the flow did separate from the airfoil, a region of low total pressure was seen above the jet. The wake survey data are contained in Appendix II.

The wake of every configuration was surveyed at the same station. But in addition, some configurations were surveyed at several stations, in order to determine the jet development. In Figure 29, the variation of the jet trajectory with the initial jet deflection angle is shown for the

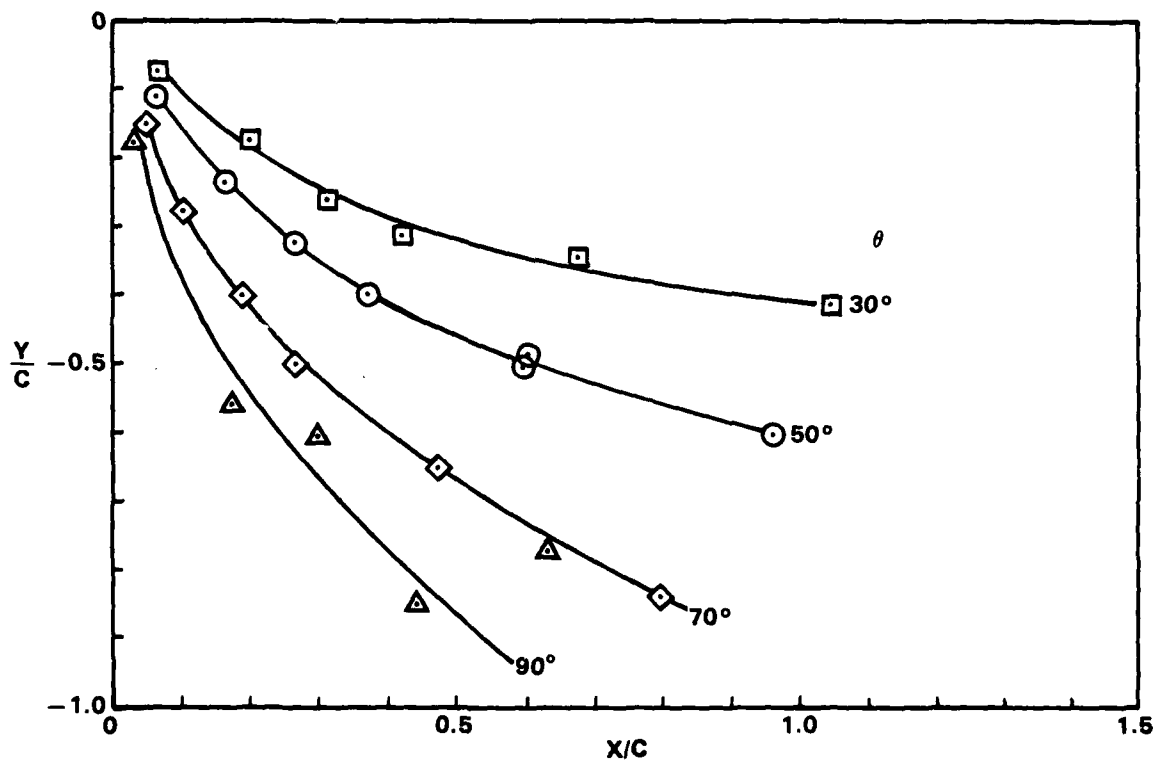


Figure 29. Variation of Jet Shape with Initial Jet Deflection Angle, θ

case with $C_\mu = 1$. As might be expected, the penetration of the jet increases with the initial deflection angle. The effect of the jet momentum coefficient on the trajectory is shown in Figure 30 for two values of the initial deflection angle. As the blowing coefficient is increased, the jet tends to straighten out. In the limit as $C_\mu \rightarrow \infty$, the jet continues along the initial trajectory and the horizontal force on the airfoil becomes $T \cos \theta$. These wake survey data are contained in Appendix III.

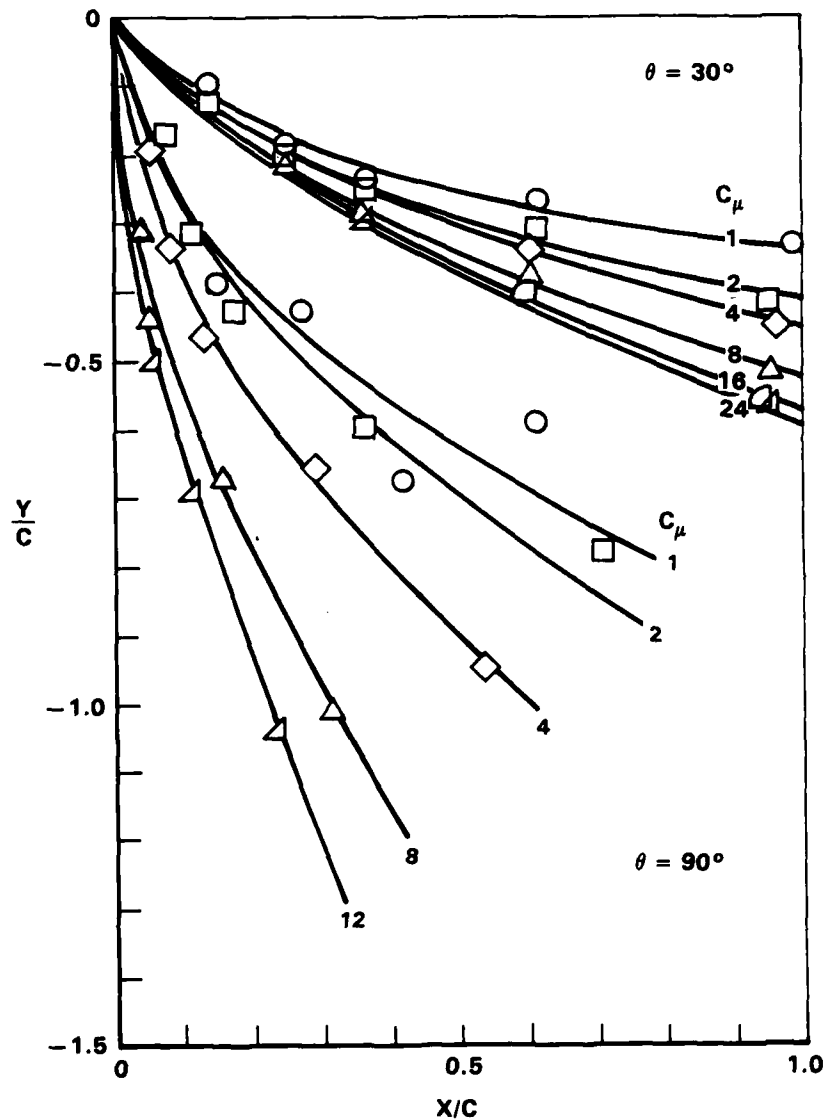


Figure 30. Variation of Jet Shape with Jet Momentum, C_μ

LIFT AND THRUST INTEGRALS

The measured surface pressure distributions were integrated to obtain the lift on the airfoil. In Figure 31 the total lift coefficient is shown as a function of the jet thrust coefficient. The total lift on the airfoil is equal to the sum of the circulation lift and the vertical reaction to the jet thrust, which appears as a suction on the flap surface. For the smaller jet deflection angles (30° , 50°) the data are well represented by the theoretical results obtained by Spence (ref 10). However, for the larger deflection angles, there is a loss of circulation lift due to separation from the airfoil. Impingement of the jet on the tunnel floor causes the lift to increase again at large values of C_μ and θ , but this is a blockage effect and not representative of the force that would be induced by an unconfined jet.

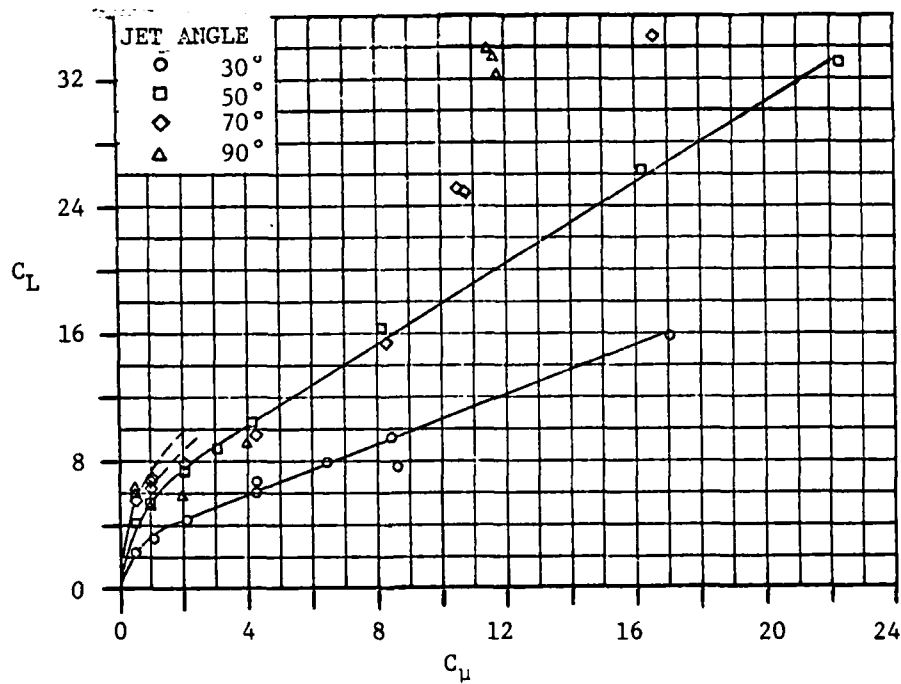


Figure 31. Lift Coefficient as a Function of Momentum Coefficient

In order to provide a check on the experimental methods, the thrust recovery was calculated from both the wake survey data and the airfoil surface pressures. As previously explained, the method of Betz (Schlichting, ref 28) was used to evaluate the excess momentum integral from the wake pressure distributions. The net thrust on the airfoil is then obtained by adding the source thrust of the initial jet mass to the momentum integral. Since the thrust recovery is defined relative to the viscous drag of the unblown airfoil, this baseline drag is added to the net thrust of the jet flap. Thus, the loss of recovery is taken to be the increase in the profile drag of the unblown airfoil due to jet entrainment and flow separation induced by the jet. The wake recovery factor is therefore given by

$$r_w = \frac{T_x + T_q + |D_v|}{T} \quad 5-2$$

in which T_x is the excess momentum integral, T_q is the source thrust, and D_v is the viscous drag.

The viscous drag of the airfoil was estimated by computing the pressure and velocity distributions on the unblown airfoil. The actual airfoil coordinates and test Reynolds number were used. Variations in lift were obtained by changing the angle of attack. In Figure 32 the predictions of this program are compared to measurements for 10° angle of attack. The excellent agreement of the measured and predicted surface pressure distributions is an indication that good two dimensionality was achieved in the wind tunnel test. The agreement between the lift coefficients is also good. The measured axial force coefficient is within 10% of the computed value; since integration of the surface pressure data to obtain the axial force is difficult, this is considered good agreement. The computed viscous drag coefficient of the airfoil is described by the function

$$C_v = .007C_L^2 + .022 \quad 5-3$$

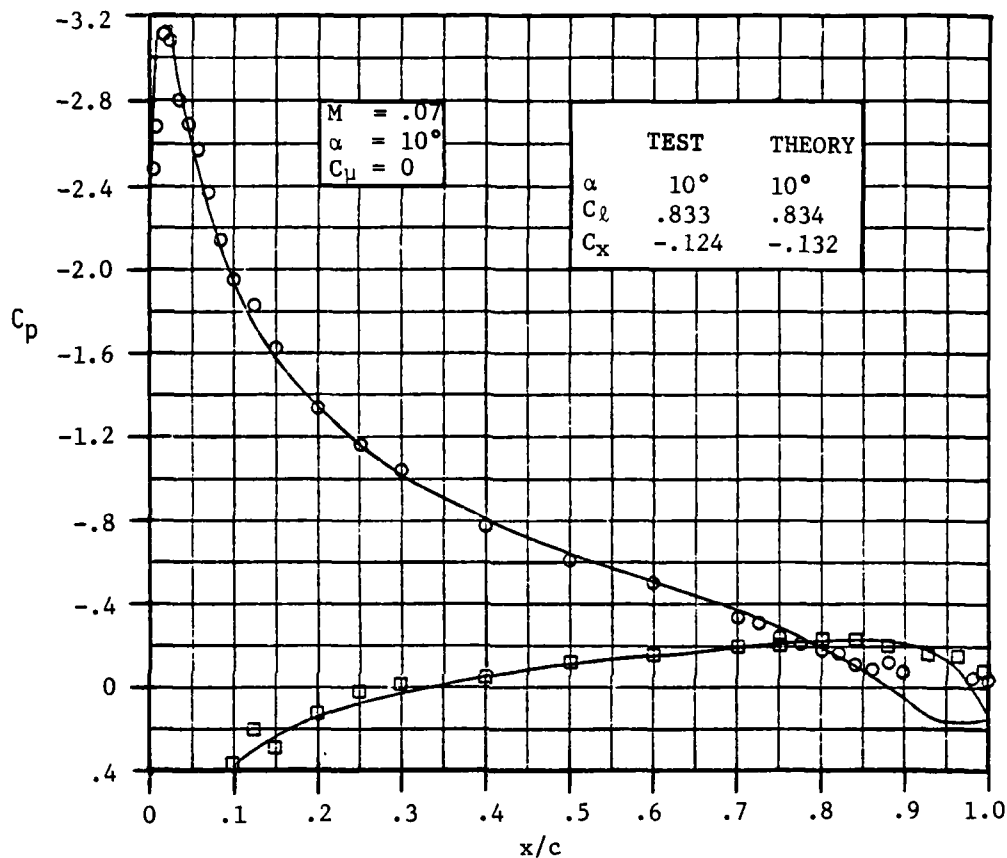


Figure 32. Comparison of Measured and Predicted Surface Pressure Distributions

in which C_L is the circulation lift coefficient. The viscous drag includes both the skin friction drag and the pressure drag due to the displacement effect of the boundary layer.

The drag obtained by integrating the airfoil surface pressures also includes the pressure drag due to viscosity, but not the skin friction

drag. Thus, the pressure drag of the unblown airfoil is added to the measured surface pressure drag. In addition, however, an induced drag correction must be made because end effects introduce some three dimensionality in the surface pressure distributions. The airfoil recovery factor is therefore given by

$$r_a = 1 - \left| \frac{D_s + D_p + D_i}{T} \right| \quad 5-4$$

in which D_s is the integral of the surface pressures, D_p is the computed viscous pressure drag of the unblown airfoil, and D_i is the induced drag correction.

Foley (ref 2) estimated the induced drag by extrapolating the linear section of the drag polar of the unblown airfoil to the large lift coefficients of the jet flapped airfoil. This is equivalent to assuming that the lift and jet thrust are elliptically distributed along the span. A similar approach was taken in this study, but the large values of the thrust coefficient considered here required using the expression for the induced drag of a jet flapped wing devised by Maskel and Spence (ref 7),

$$C_{Di} = \frac{C_L^2}{\pi A Re + 2C_\mu} \quad 5-5$$

In effect, Foley (ref 2) neglected the thrust coefficient term, but over the range of values he examined ($C_\mu < 1$), the error is small. In this study, the effective aspect ratio of the airfoil was determined by subtracting the computed viscous drag from the measured total drag of the unblown airfoil. The measured drag polar is shown in Figure 33. The total drag is described by the function

$$C_D = .011C_L^2 + .022 \quad 5-6$$

The effective aspect ratio of the airfoil is then obtained from the relation

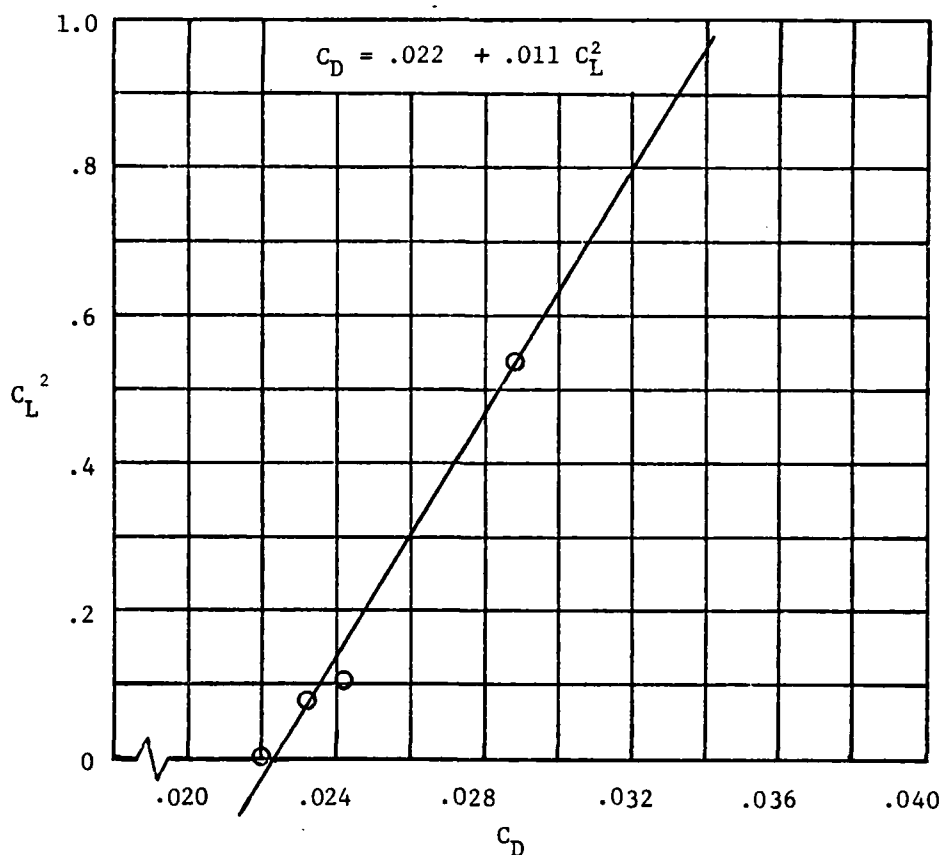


Figure 33. Drag Polar of the Unblown Airfoil

$$ARe = \frac{C_L^2}{\pi(C_D - C_v)}$$

5-7

This gives $ARe = 77$, which was then assumed to be independent of variations in C_μ and θ_0 .

The values of the thrust recovery factor obtained by both methods are compared in Figure 34. There is surprisingly good agreement between the methods, although the surface pressure data gives consistently higher

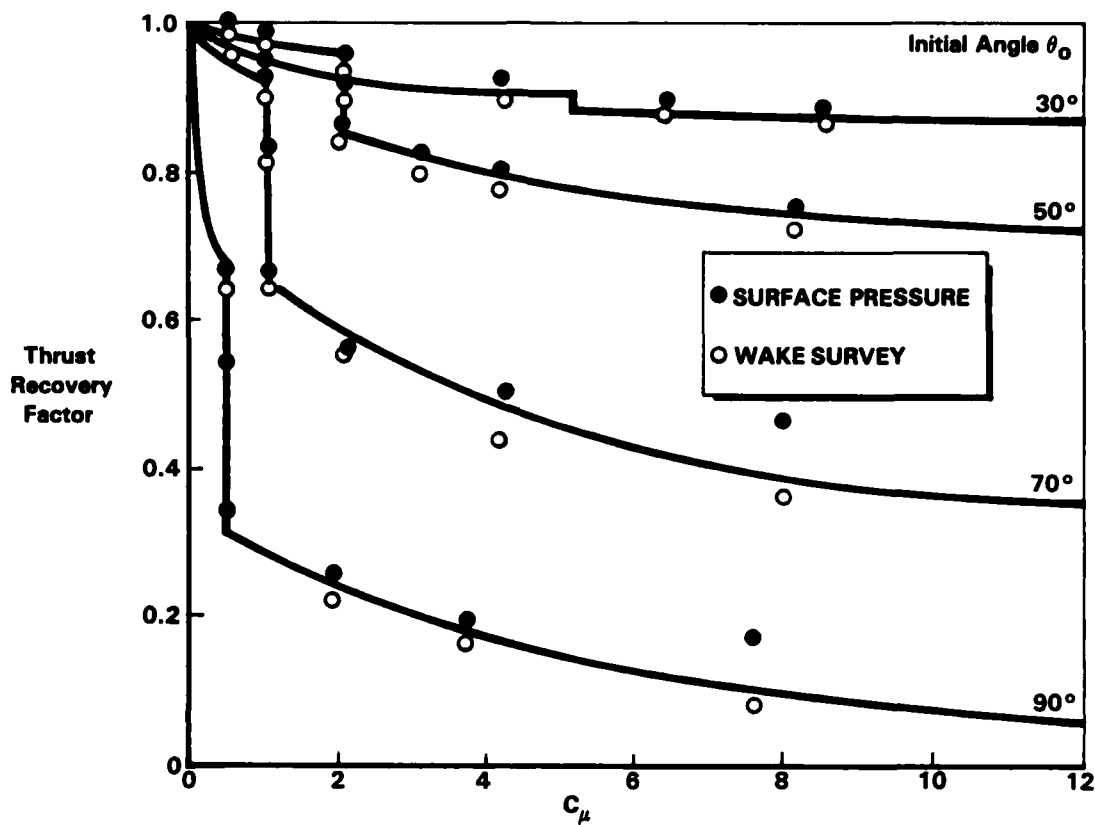


Figure 34. Variation of the Thrust Recovery Factor

recovery. Based on our prior experience with wake and surface pressure integrations, it is our feeling that the wake survey data is more reliable. However, the differences observed are within the 16% range of experimental error associated with previous measurements of thrust recovery (Leaman and Plotkin, ref 29). The increase in the recovery factor observed for large values of C_μ and θ_0 is due to jet impingement on the tunnel floor, and is not representative of the thrust recovery of an unconfined jet.

ANALYSIS OF RESULTS

In general, the thrust recovery decreases as the jet thrust coefficient and initial deflection angle are increased. However, the variation is not

continuous. For every value of the deflection angle there is a discontinuous change in the recovery for a value of the thrust coefficient of approximately $C_{\mu} = 2$. During the test, a distinct increase in noise level could be heard when the discontinuity occurred. Comparison of the measured airfoil surface pressure distributions just before and after the discontinuity reveals that it is due to the bursting of the leading edge separation bubble. This is shown in Figure 35 for the $\theta_0 = 50^\circ$ case. There is a change between a short separation bubble, which gives larger recovery and a long bubble which gives less recovery. Although the thrust recovery continues to decrease as the long bubble grows, there is no change when the bubble ultimately fails to reattach to the surface of

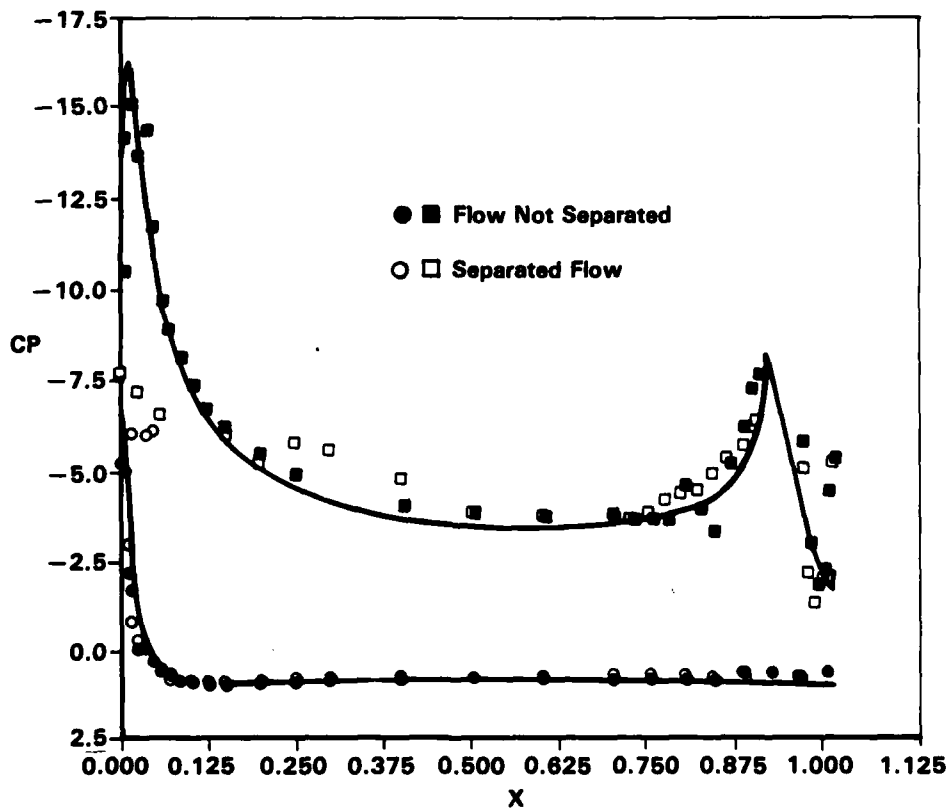


Figure 35. Comparison of Measured and Ideal Airfoil Pressure Distributions, $C_{\mu} = 2$, $\theta = 50^\circ$

the airfoil at all. Because earlier studies were conducted at small values of C_μ , this discontinuity has not been previously known.

The bursting of the bubble depends on the Reynolds number and Mach number of the flow, and the leading edge radius of the airfoil, so that these trends cannot be generally applied. However, the value of the thrust recovery factor is well predicted by the jet flap analysis up to the point of bursting. Thus, it can be concluded that the loss of recovery up to the burst point is due to jet entrainment drag, and that the short bubble does not significantly affect either the lift or thrust integrals.

In Figure 36 the computed surface pressure distribution on the unblown airfoil is compared to the computed distribution on an airfoil with an

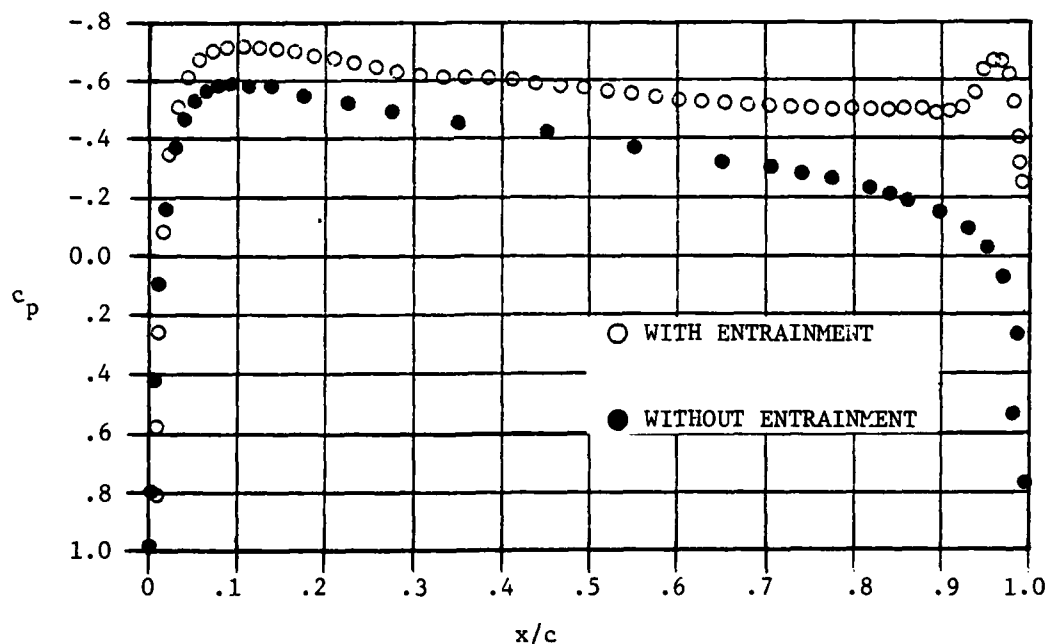


Figure 36. Comparison of Airfoil Surface Pressure Distributions

undeflected jet, for $C_{\mu} = 1$. The effect of the jet is to increase the base drag on the airfoil. Integration of the surface pressures gives $r = 0.050$ for the thrust recovery. The corresponding variation in the excess momentum of the jet is shown in Figure 37. By the last station shown, the jet thrust has decreased to give $r = 0.051$.

For small values of the jet thrust coefficient, the thrust recovery actually increases slightly as the jet deflection angle is increased from 30° to 50° . Similar behavior was seen in Foley's (ref 2) data, and predicted by the jet flap analysis. Because the static pressure on the upper surface of the jet becomes more negative as the jet is deflected, there may actually be some thrust augmentation in this case. However, the effect is small and not sufficient to overcome the entrainment drag which occurs farther downstream.

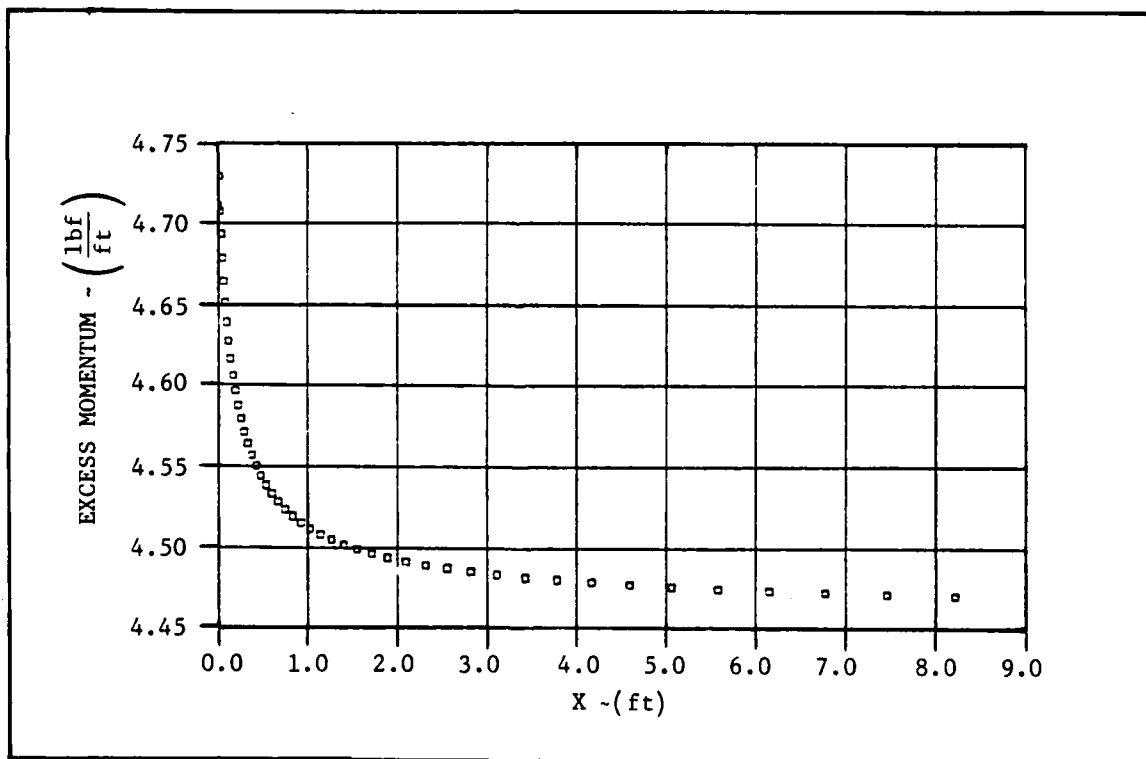


Figure 37. Streamwise Variation of Excess Momentum in Jet

VI. CONCLUSIONS AND RECOMMENDATIONS

The thrust recovery is generated by equal but opposite pressure forces induced on the airfoil and the jet flap. If the mixing of the jet is neglected, the jet is a streamline of the flow, and the net pressure force is everywhere normal to the jet. In this case, the thrust of the jet is conserved and ultimately recovered as a horizontal force in the far wake. This gives complete thrust recovery. Real fluid effects reduce the actual recovery both by inducing a jet drag on the airfoil and by causing separation of flow from the airfoil.

The thrust recovery is a function of both the thrust coefficient and jet deflection angle. In general, the recovery is nearly complete for small values of the thrust coefficient ($C_{\mu} < 1$), but decreases to the horizontal component of the jet reaction force at large values of thrust ($C_{\mu} > 15$). The effect of jet mixing is to reduce the thrust recovery about 10% at small values of the thrust coefficient and deflection angle. As these parameters are increased, flow separation causes a further loss in recovery. For a fixed deflection angle, there is a discontinuous change in the recovery factor at $C_{\mu} \sim 2$, as the character of the separation changes from a "short" bubble which reattaches to the airfoil near the leading edge, to a "long" bubble which reattaches near the trailing edge. The flow is bistable in the transition region. At higher values of C_{μ} , the bubble does not reattach to the airfoil at all, but is entrained into the wake.

A viscid/inviscid interaction analysis has been developed to compute the contribution of the jet drag to the reduction in thrust recovery. It was found that the entrainment drag accounts for all the variation in recovery before bursting occurs. Thus, such an analysis can be used to predict the thrust recovery of various jet flap concepts if the flow is prevented from separating by the use of slats, etc.

Because flow separation plays such an important part in determining the thrust recovery at large values of the thrust coefficient, further development of jet flap theory should focus on predicting the effects of separation.

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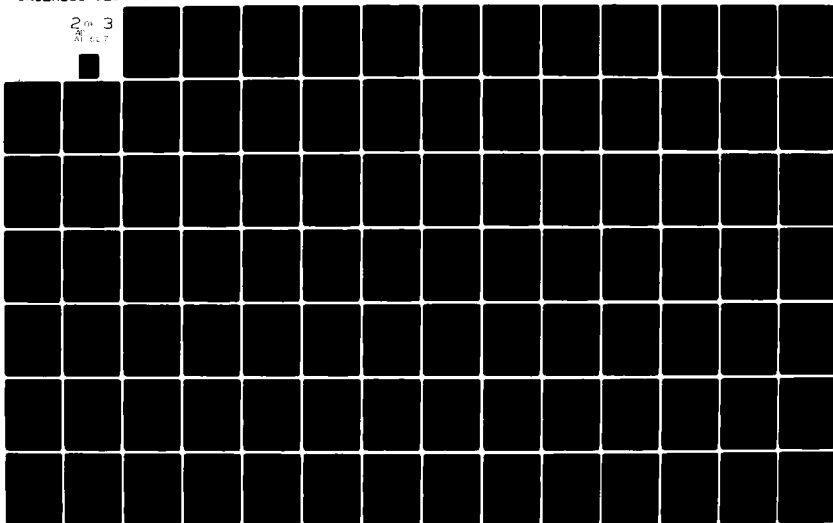
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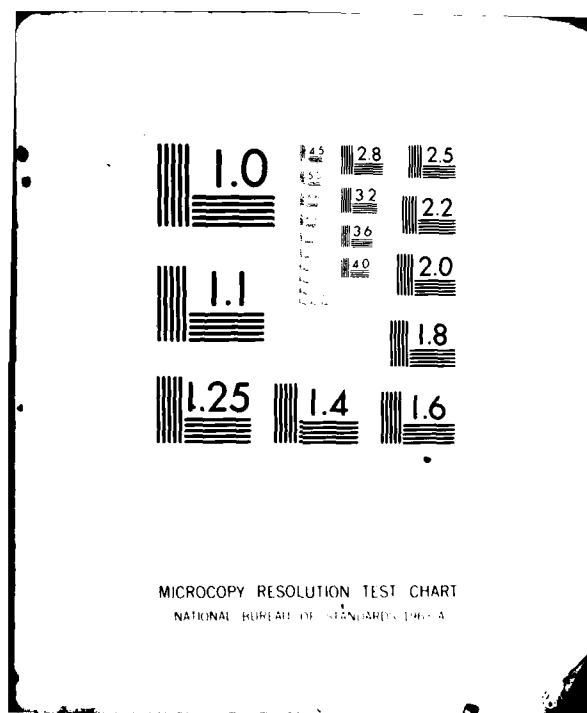
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PROGRESS TOWARDS A THEORY OF JET FLAP THRUST RECOVERY.(U)

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APPENDIX I

AIRFOIL SURFACE PRESSURE DISTRIBUTION

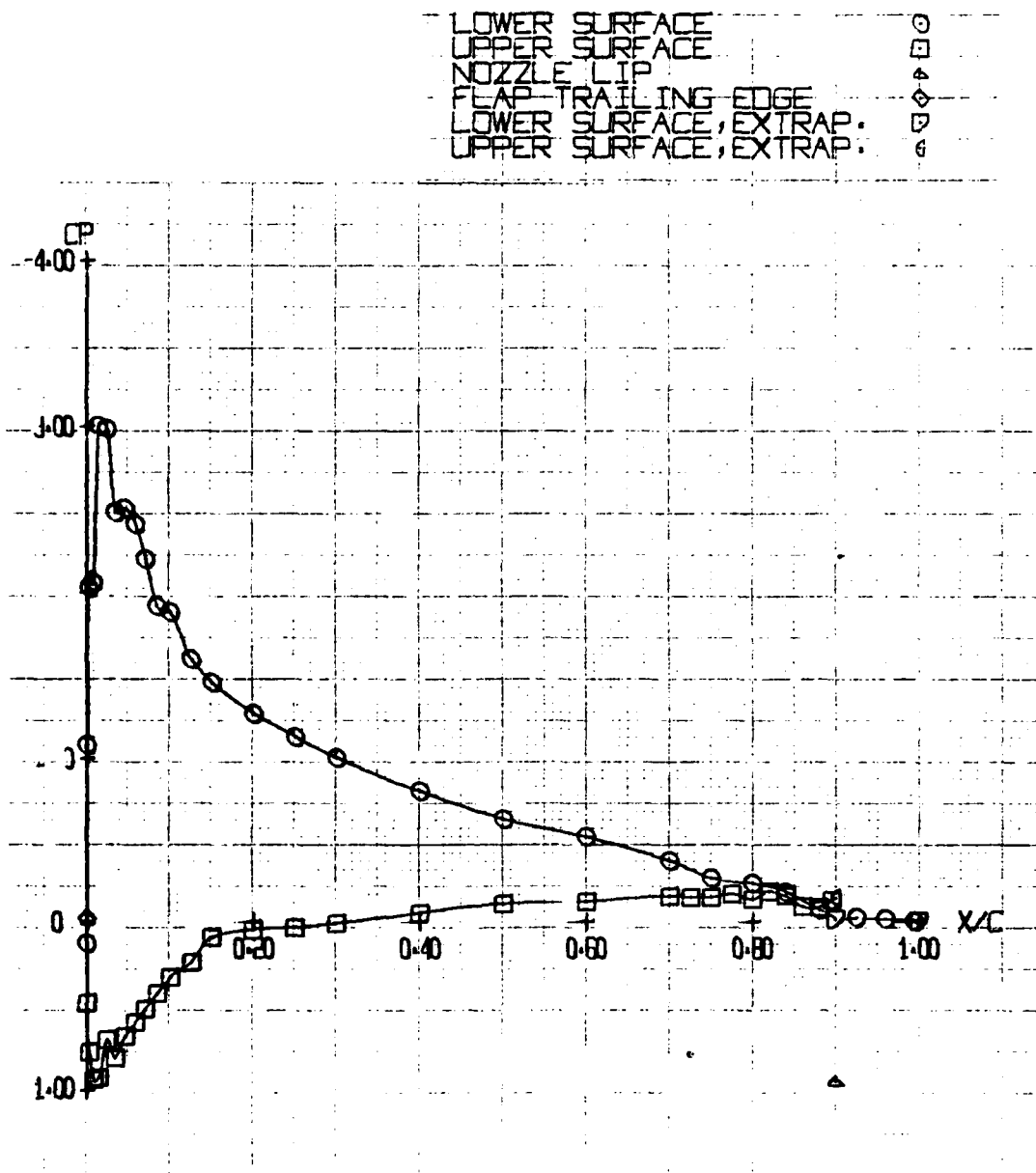


Figure I-1. $C_u = 0$, $\Theta = 0^\circ$, $\alpha = -10^\circ$ (Test 272, Run 2)
 $Q = 7.18$ psf

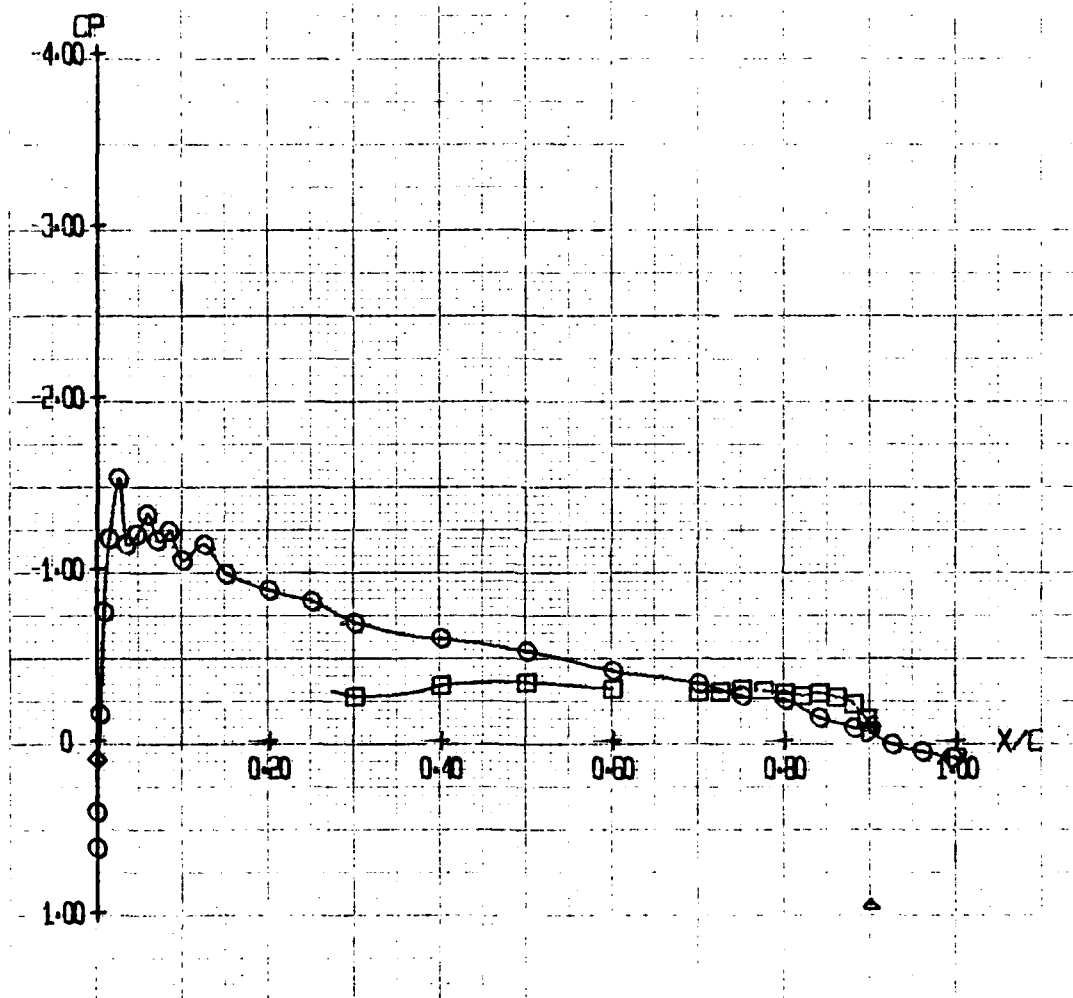


Figure I-2. $C_\mu = 0$, $\Theta = 0^\circ$, $\alpha = -5^\circ$ (Test 272, Run 3)
 $Q = 7.28$ psf

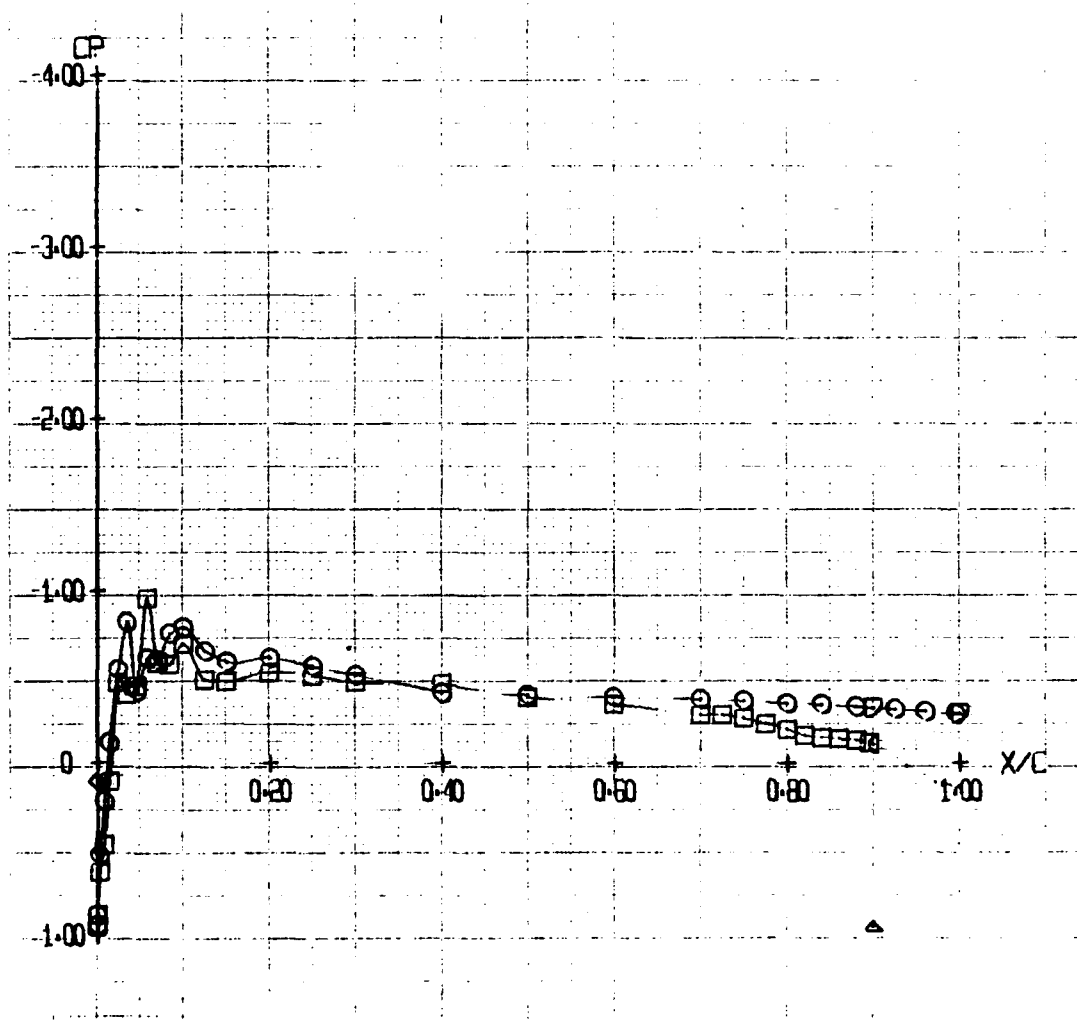


Figure I-3. $C_\mu = 0^\circ$, $\Theta = 0^\circ$, $\alpha = 0^\circ$ (Test 272, Run 1)
 $Q = 7.27$ psf

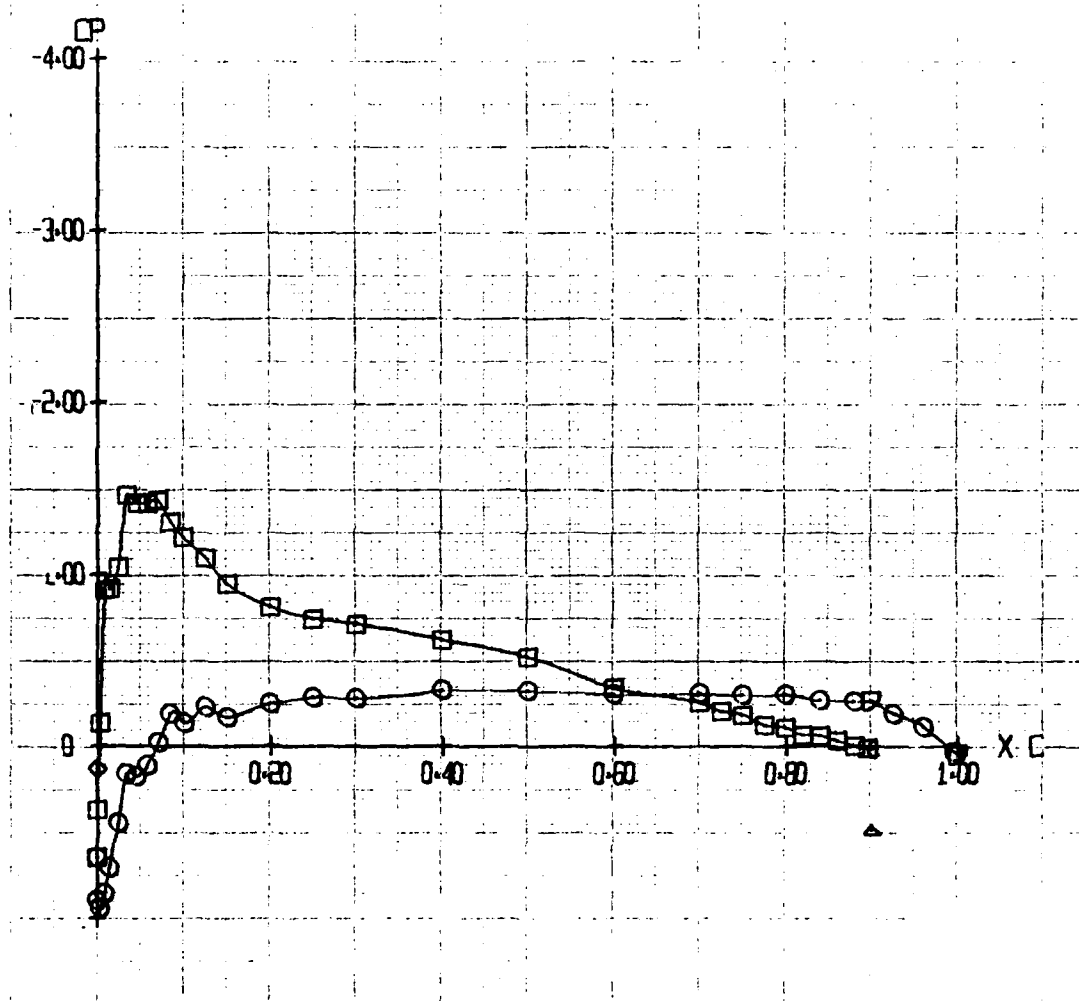


Figure I-4. $C_{\mu} = 0$, $\Theta = 0^\circ$, $\alpha = 0^\circ$ (Test 272, Run 4)
 $Q = 7.25$ psf

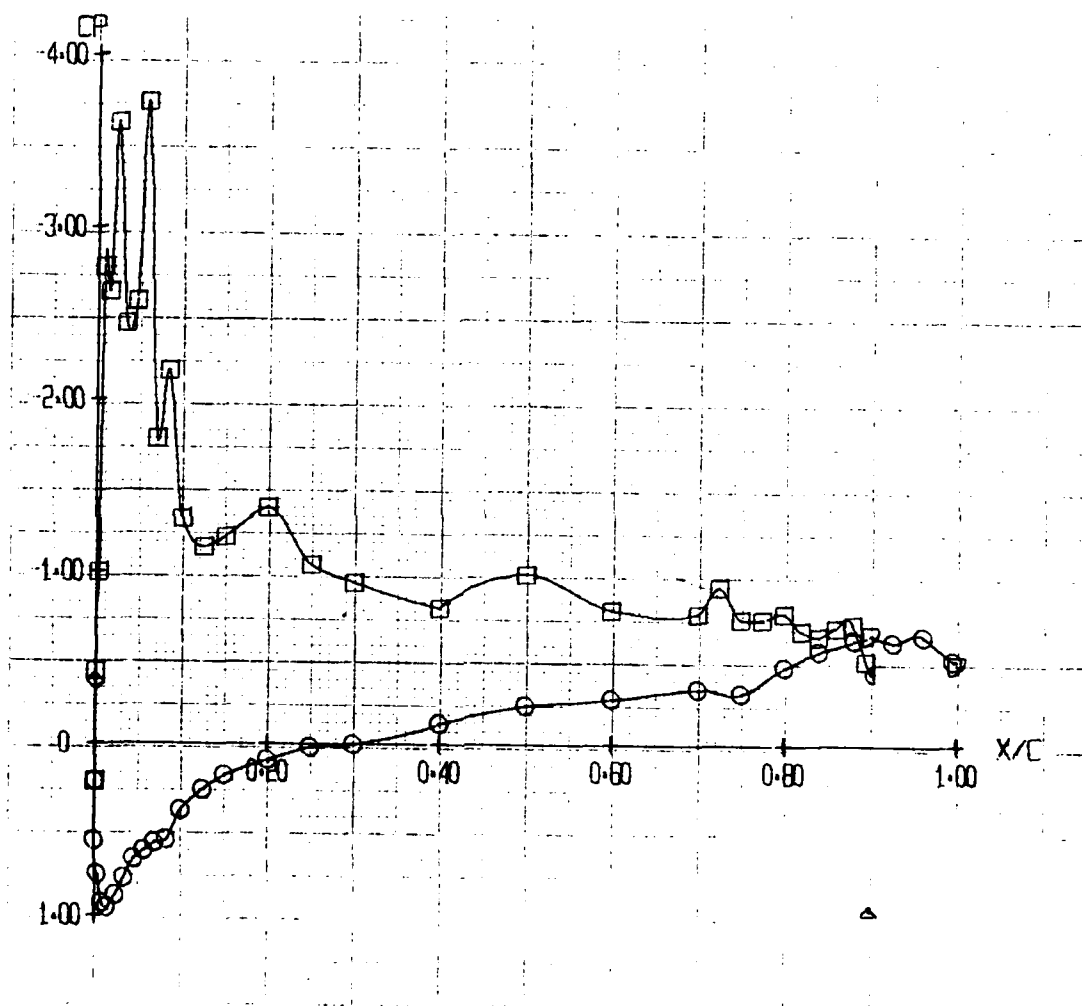


Figure I-5. $C_{\mu} = 0$, $\Theta = 0^\circ$, $\alpha = 15^\circ$ (Test 272, Run 8)
 $Q = 7.22$ psf

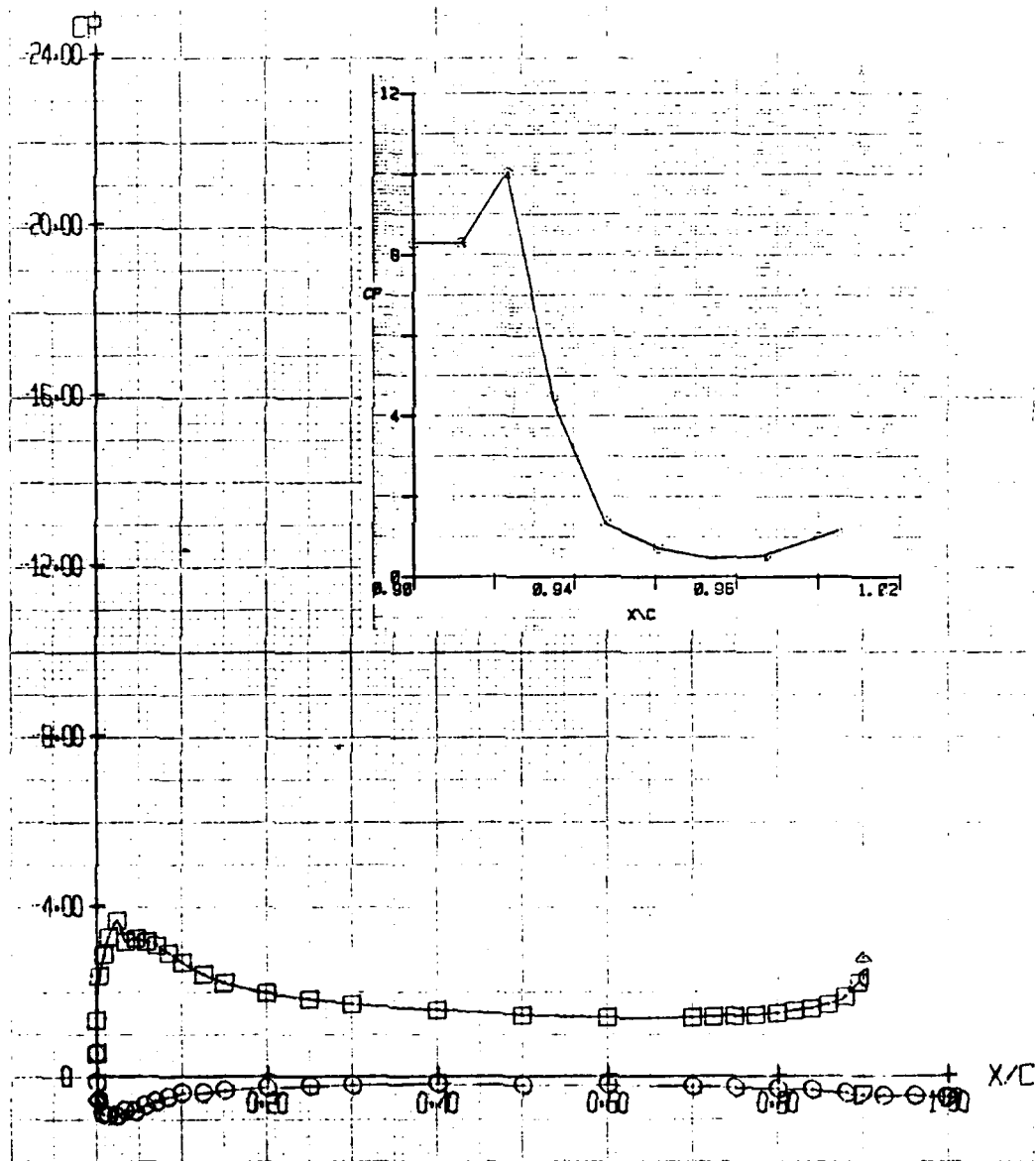


Figure I-6. $C_u = 0.53$, $\Theta = 30^\circ$, $\alpha = 0^\circ$ (Test 272, Run 9)
 $Q = 6.91$ psf

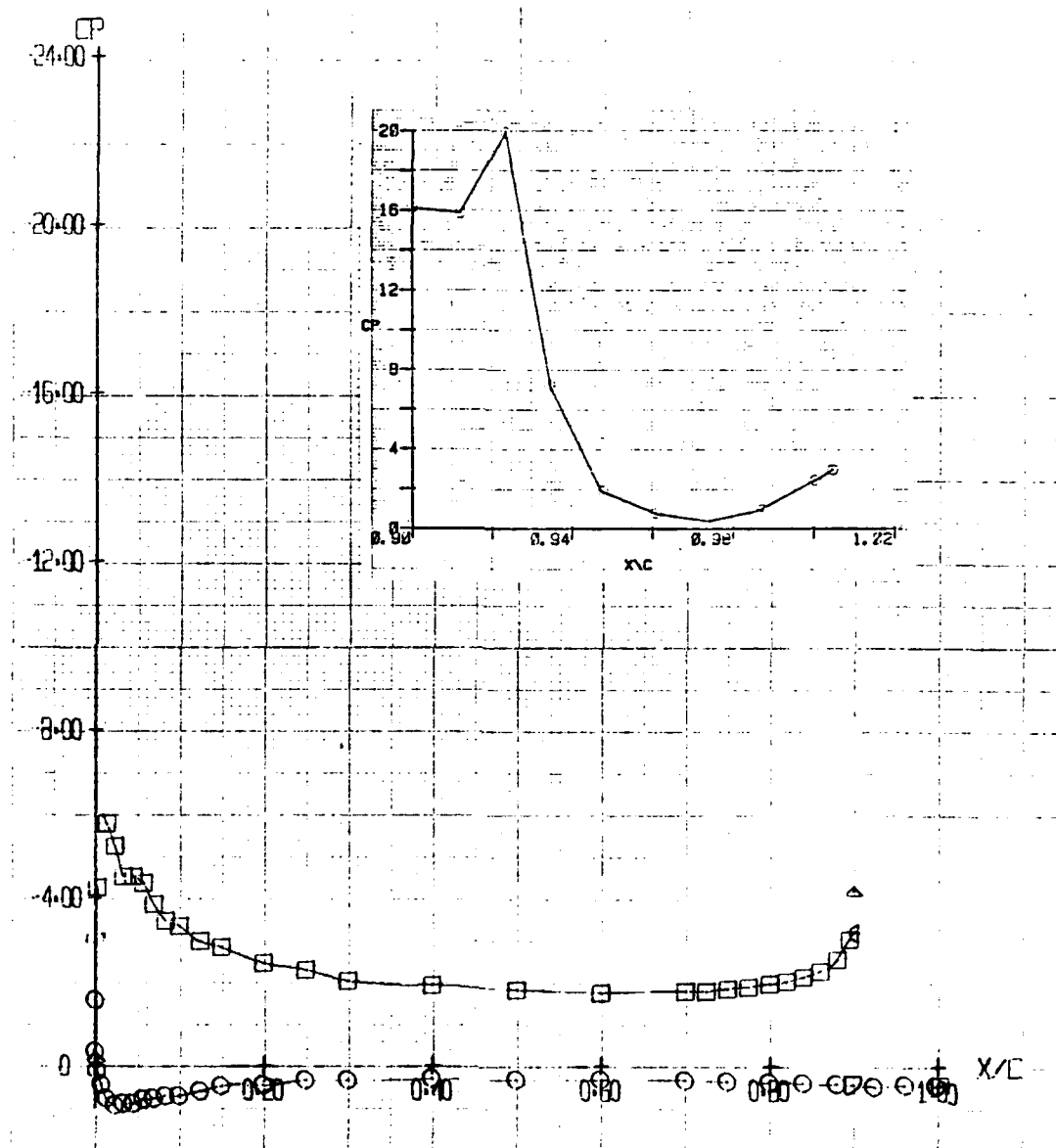


Figure I-7. $C_\mu = 1.07$, $\Theta = 30^\circ$, $\alpha = 0^\circ$ (Test 272, Run 10)
 $Q = 7.07$ psf

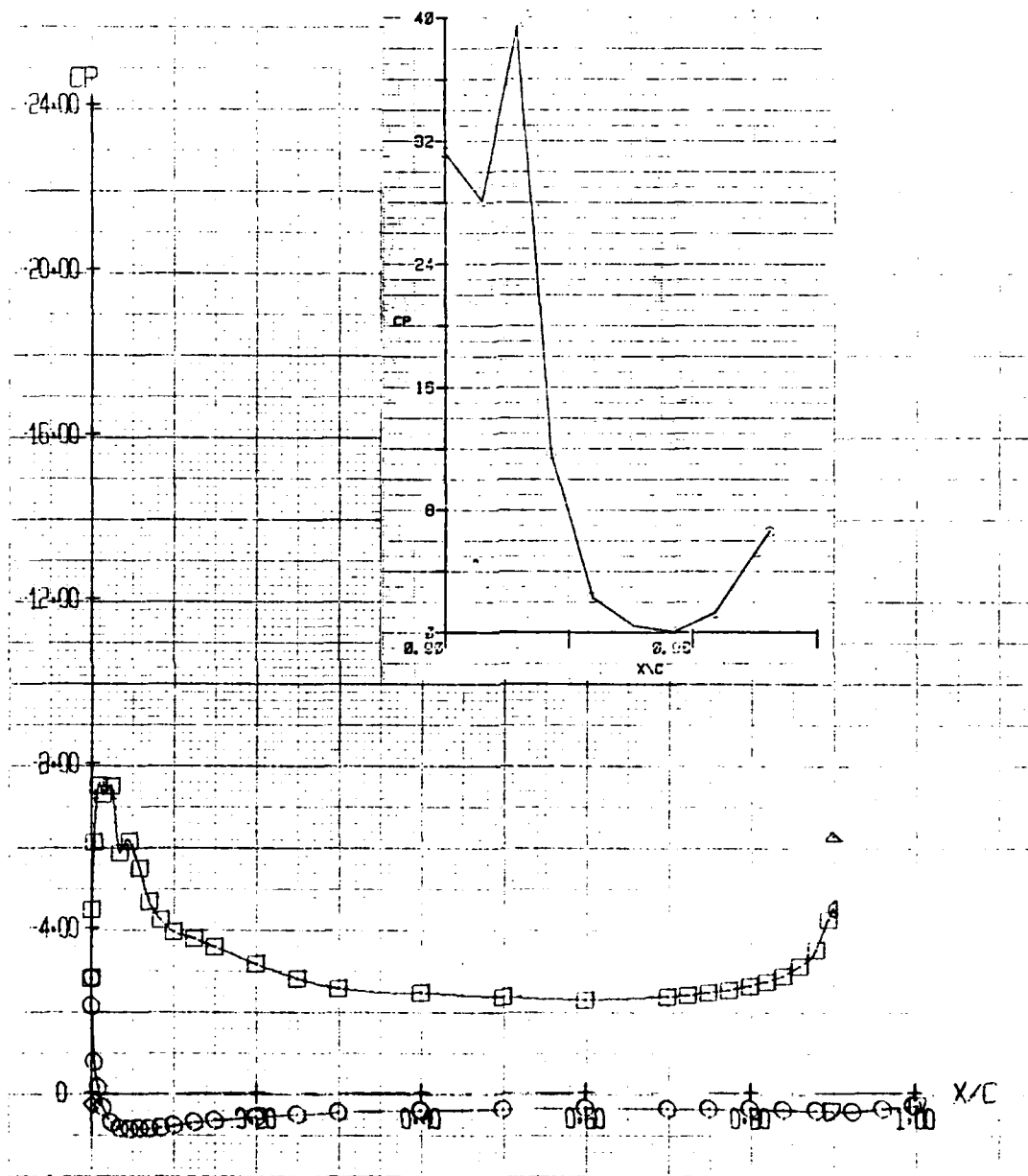


Figure I-8. $C_\mu = 2.11$, $\Theta = 30^\circ$, $\alpha = 0^\circ$ (Test 272, Run 11)
 $Q_\mu = 7.16$ psf

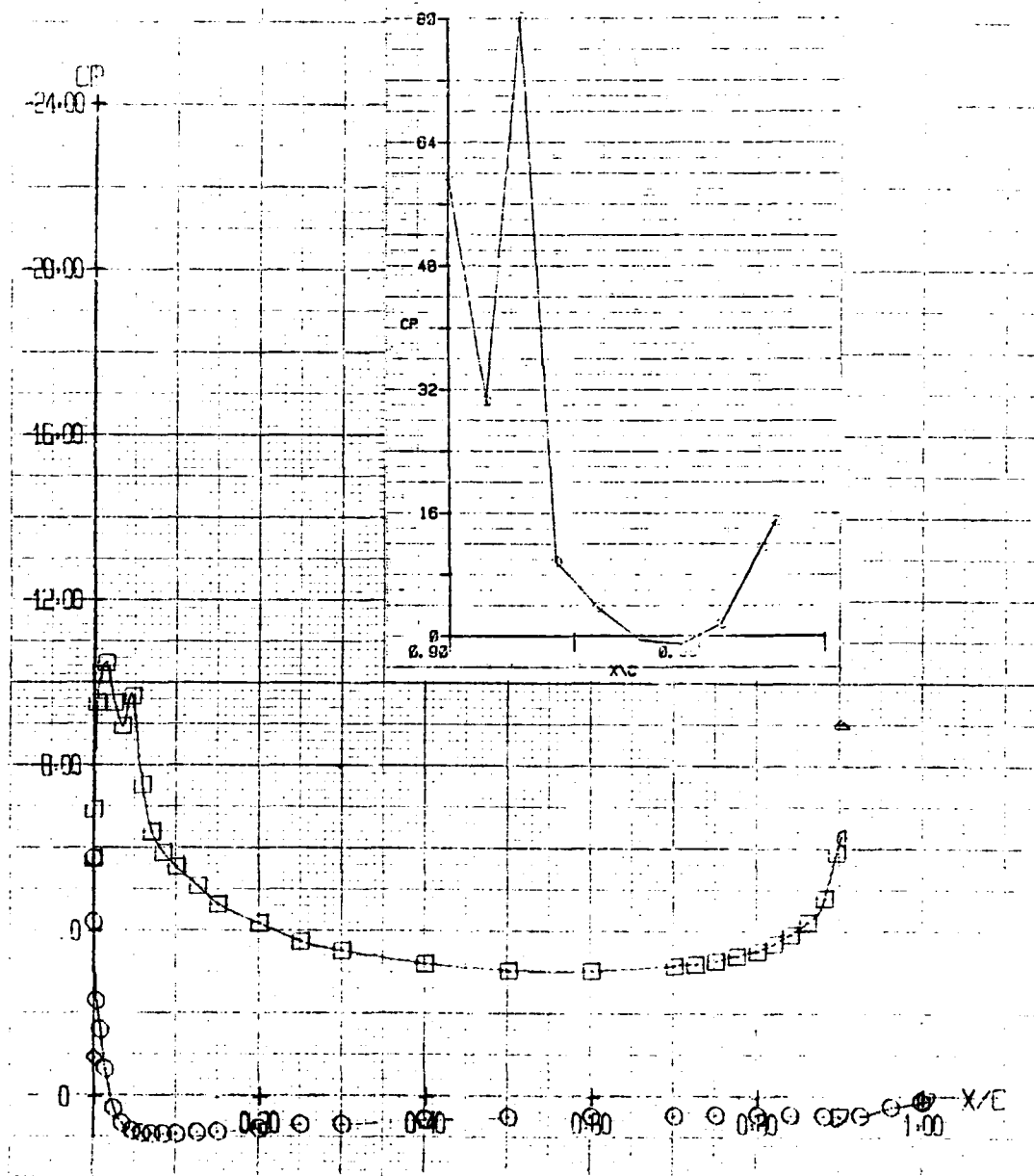


Figure I-9. $C_{\mu} = 4.26$, $\Theta = 30^\circ$, $\alpha = 0^\circ$ (Test 272, Run 13)
 $Q = 6.13$ psf

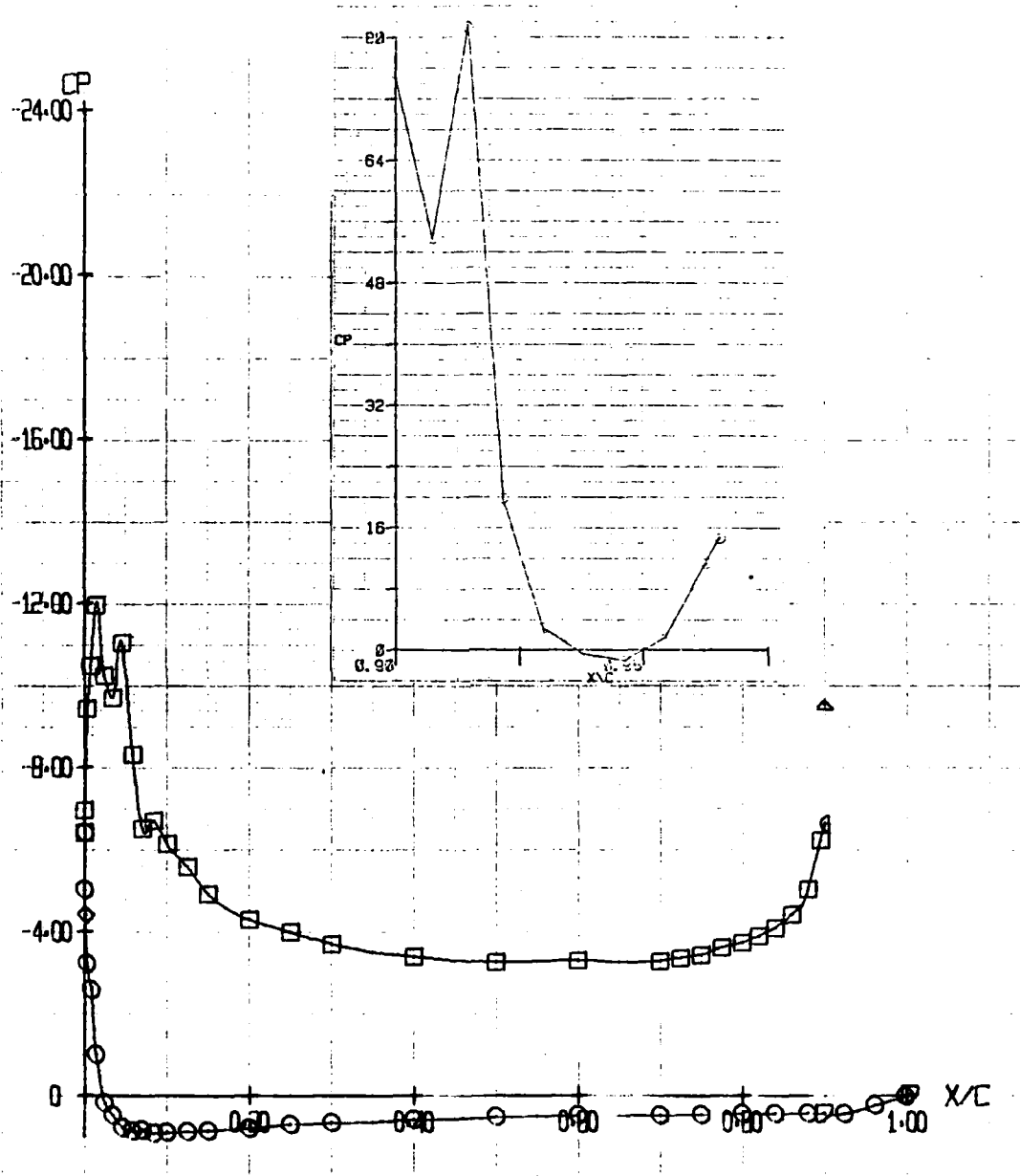


Figure I-10. $C_u = 4.26$, $\Theta = 30^\circ$, $\alpha = 0^\circ$ (Test 272, Run 14)
 $Q = 4.52$ psf

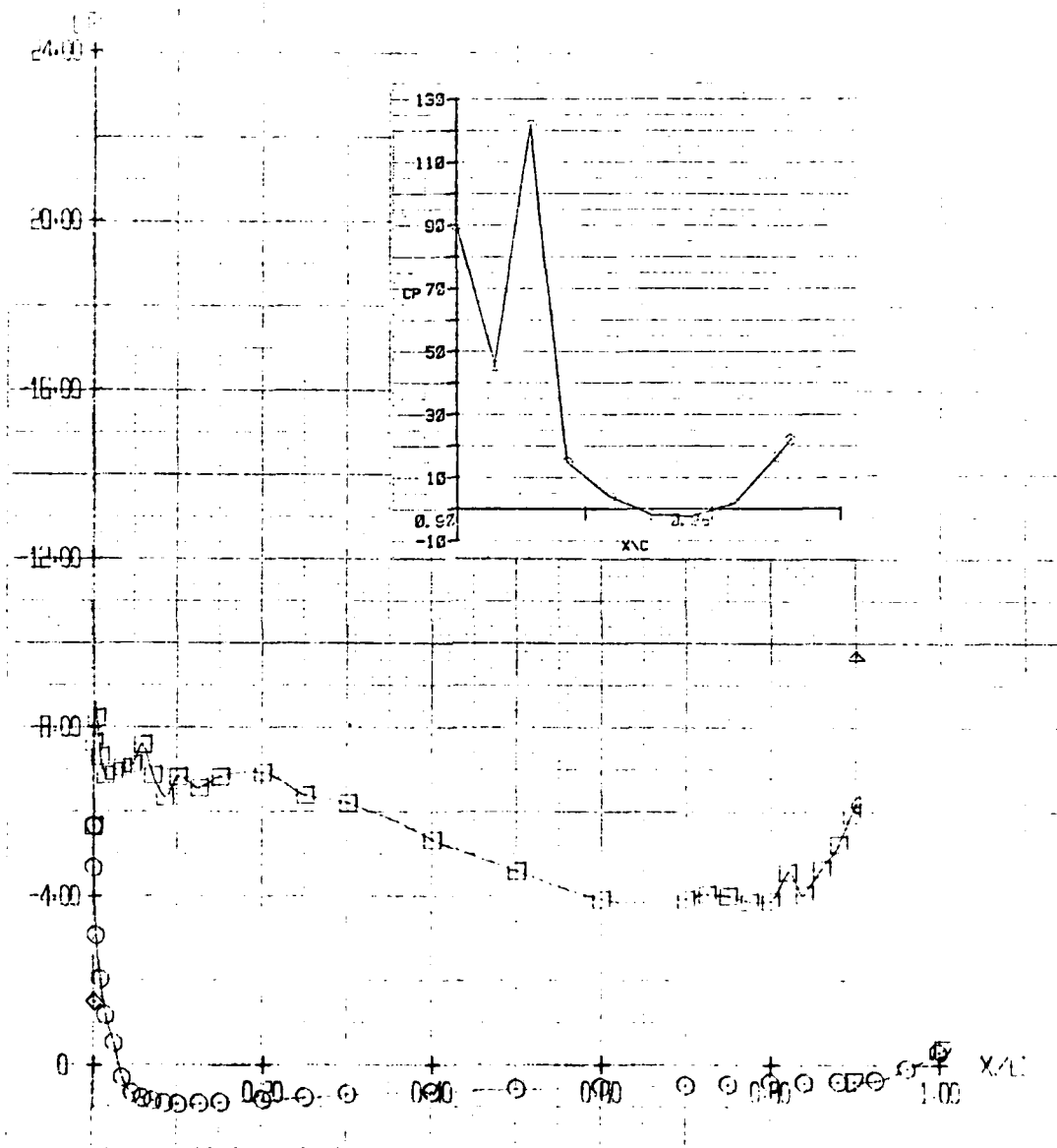


Figure I-11. $C_u = 6.40$, $\Theta = 30^\circ$, $\alpha = 0^\circ$ (Test 272, Run 15)
 $Q = 4.04$ psf

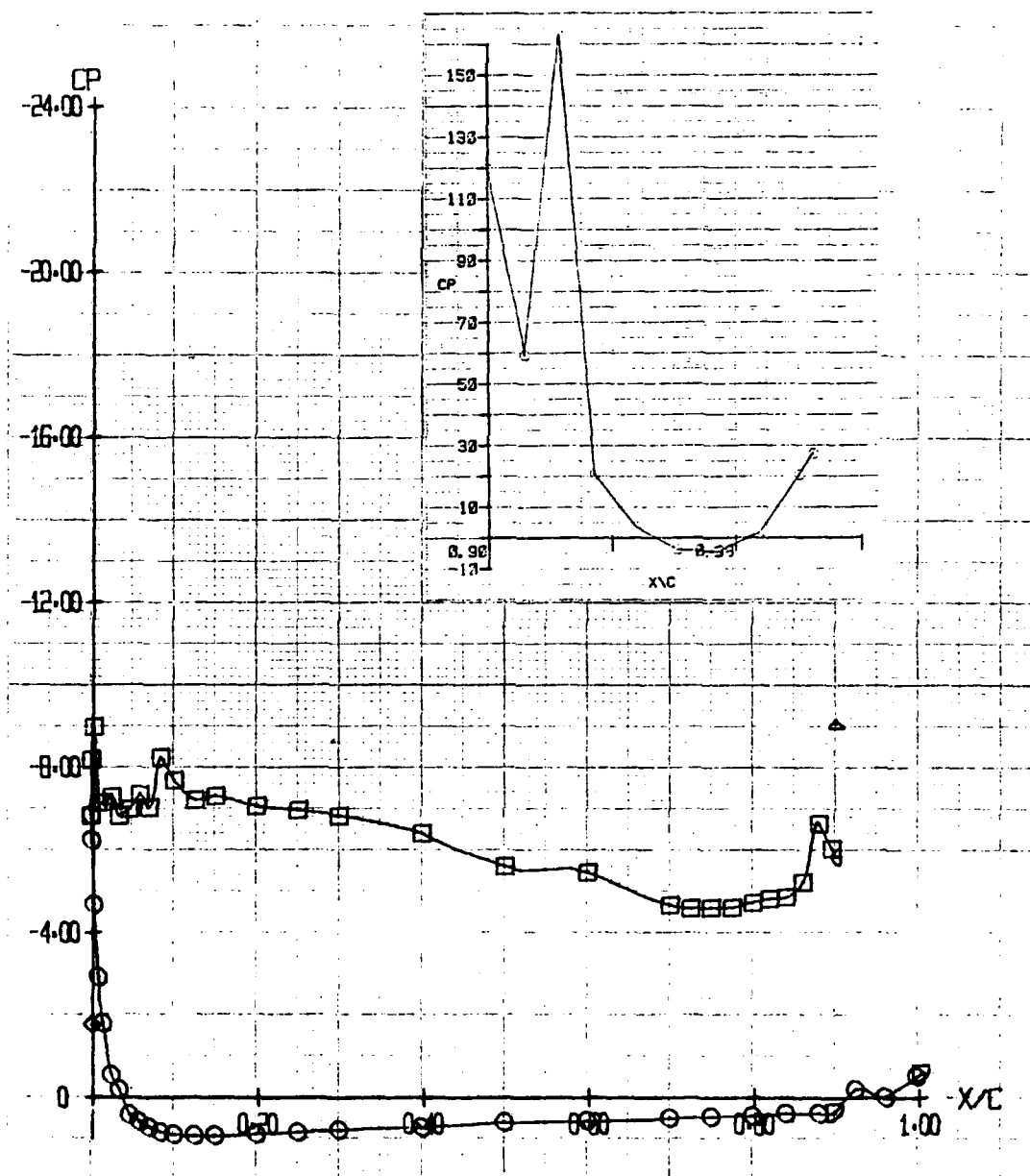


Figure I-12. $C_u = 8.43$, $\Theta = 30^\circ$, $\alpha = 0^\circ$ (Test 272, Run 16)
 $Q = 3.04$ psf

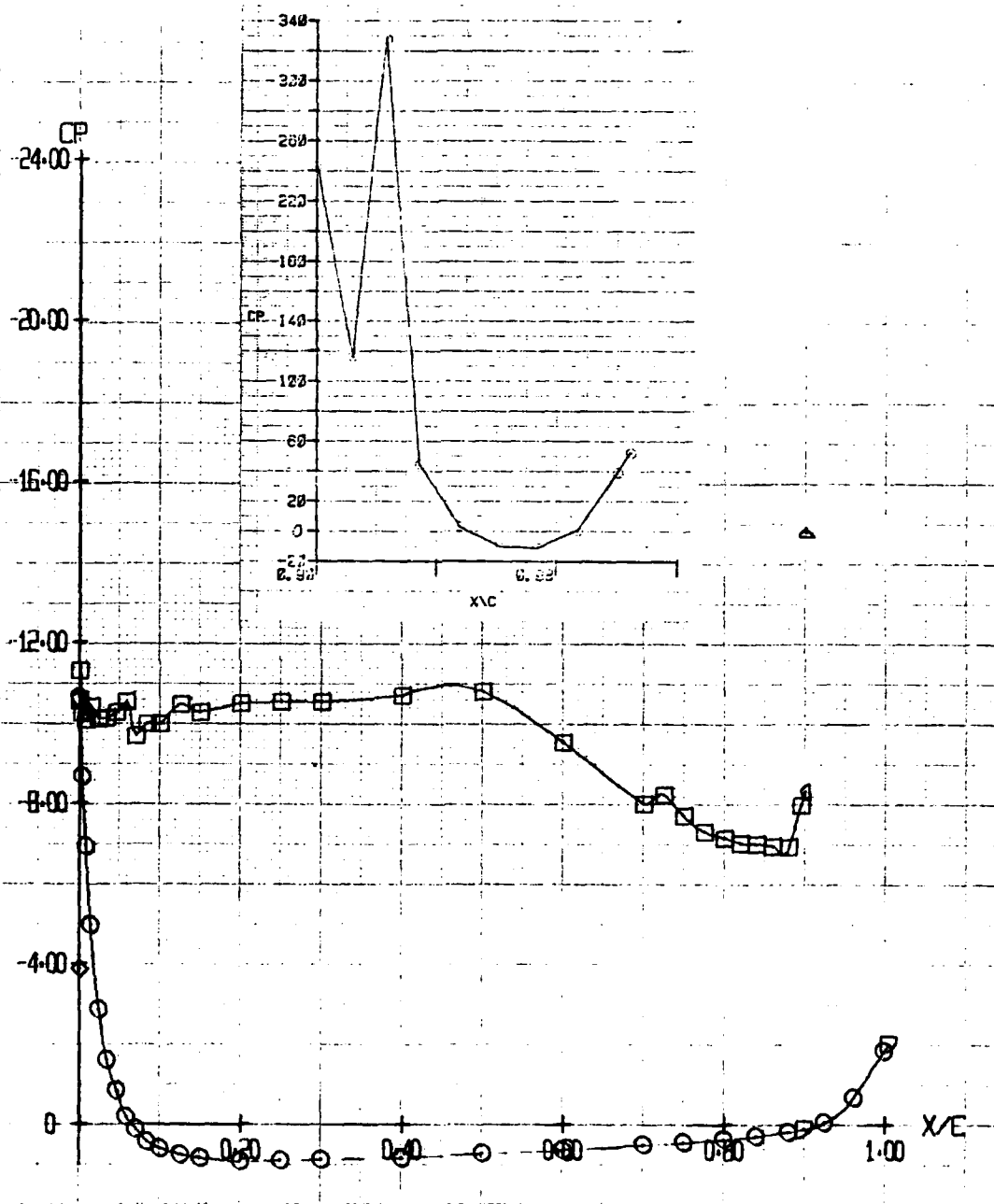


Figure I-13. $C_u = 17.03$, $\Theta = 30^\circ$, $\alpha = 0^\circ$ (Test 272, Run 18)
 $Q = 1.51$ psf

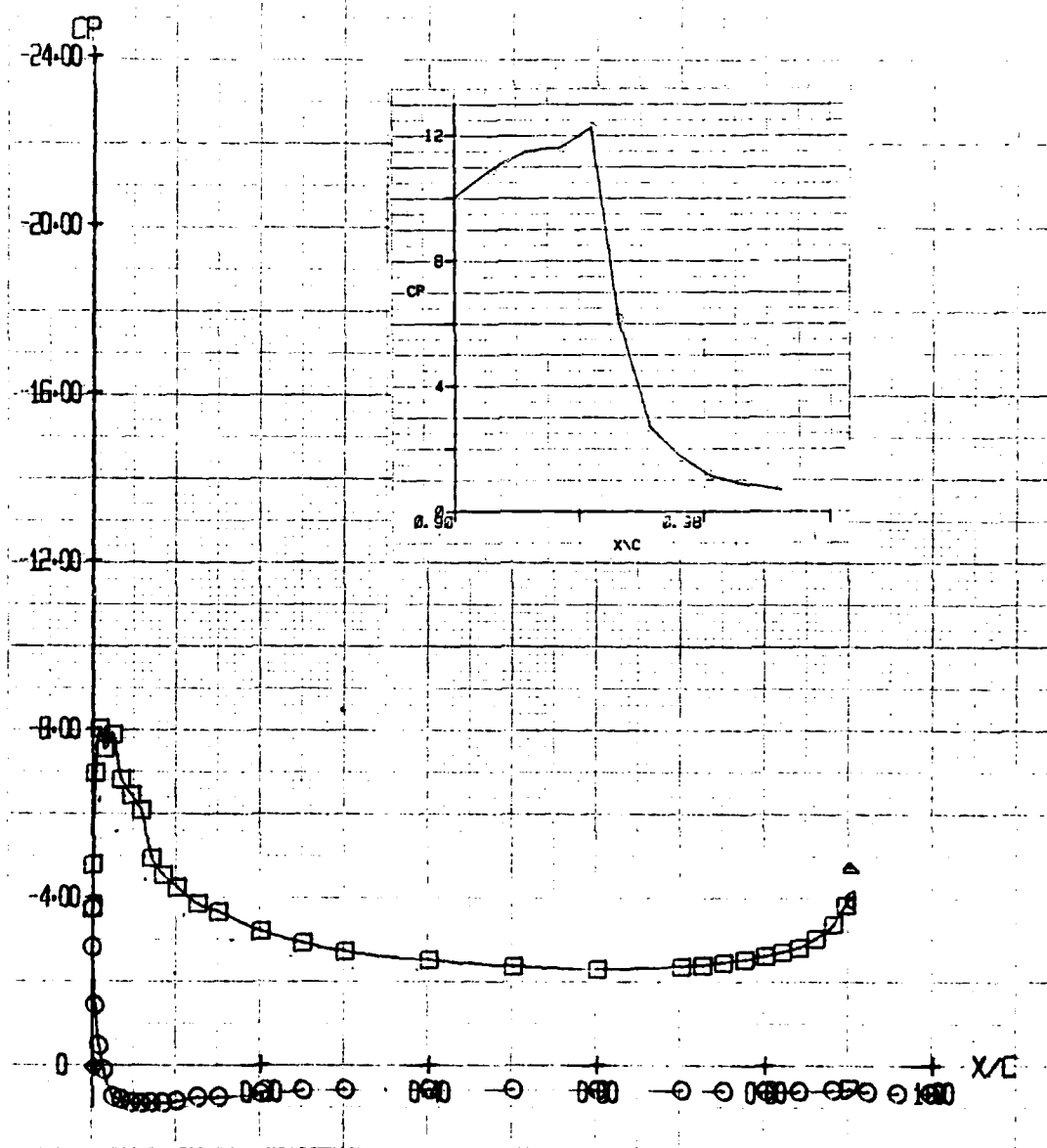


Figure I-14. $C_u = 0.52$, $\theta = 50^\circ$, $\alpha = 0^\circ$ (Test 272, Run 19)
 $Q = 7.14$ psf

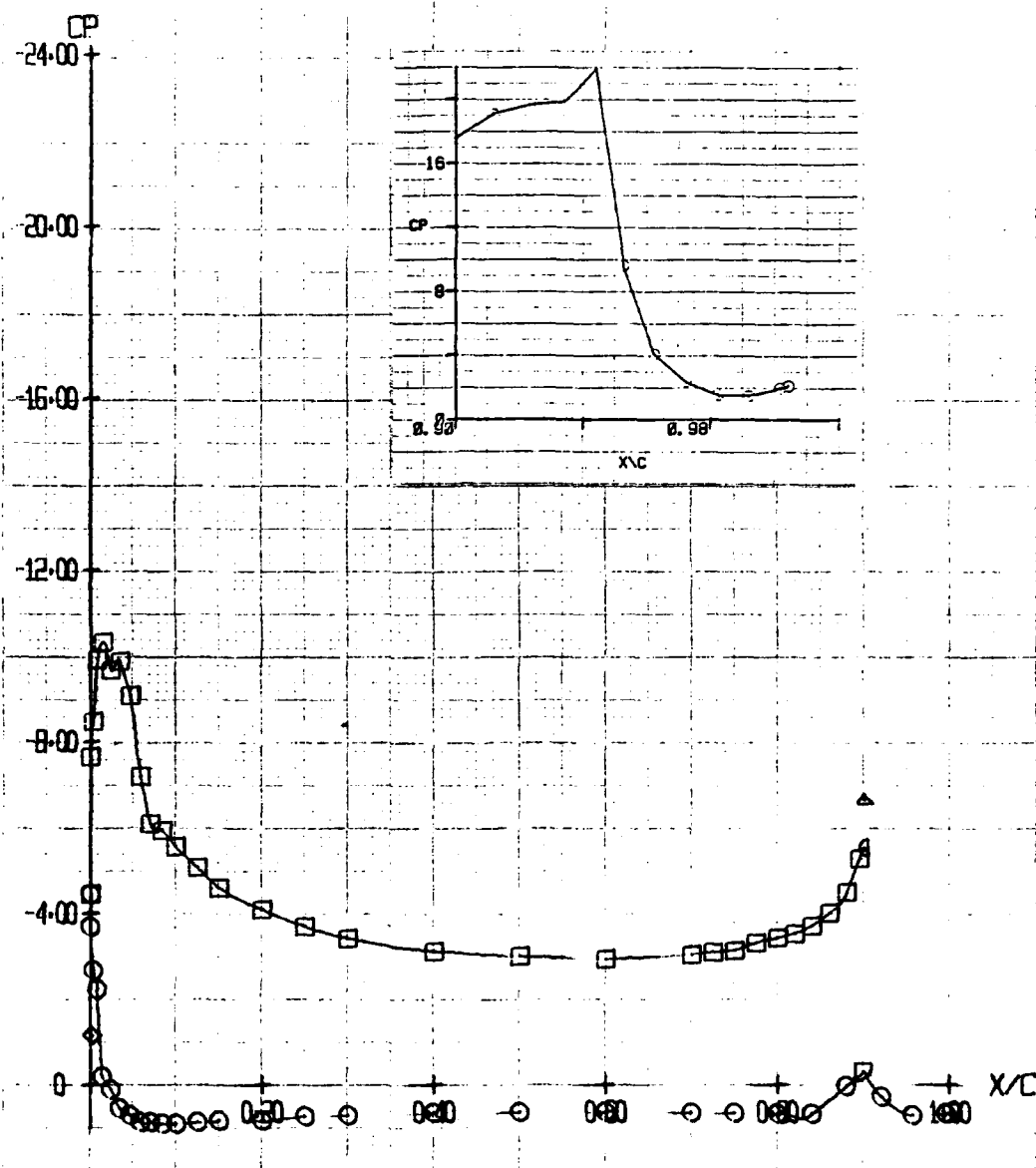


Figure I-15. $C_u = 1.01$, $\Theta = 50^\circ$, $\alpha = 0^\circ$ (Test 272, Run 20)
 $Q = 7.07$ psf

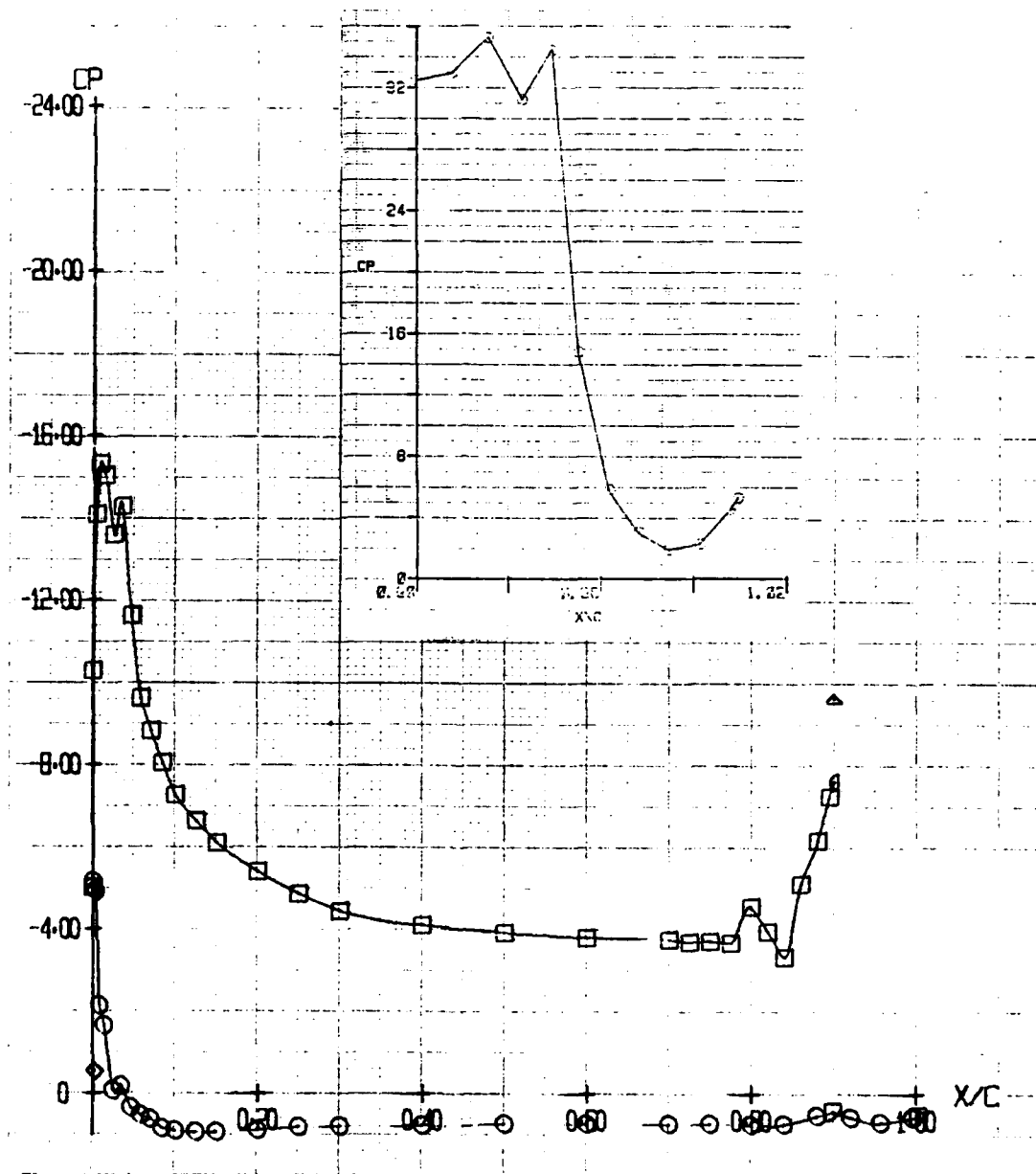


Figure I-16. $C_{\mu} = 2.05$, $\Theta = 50^\circ$, $\alpha = 0^\circ$ (Test 272, Run 24)
 $Q = 7.02$ psf

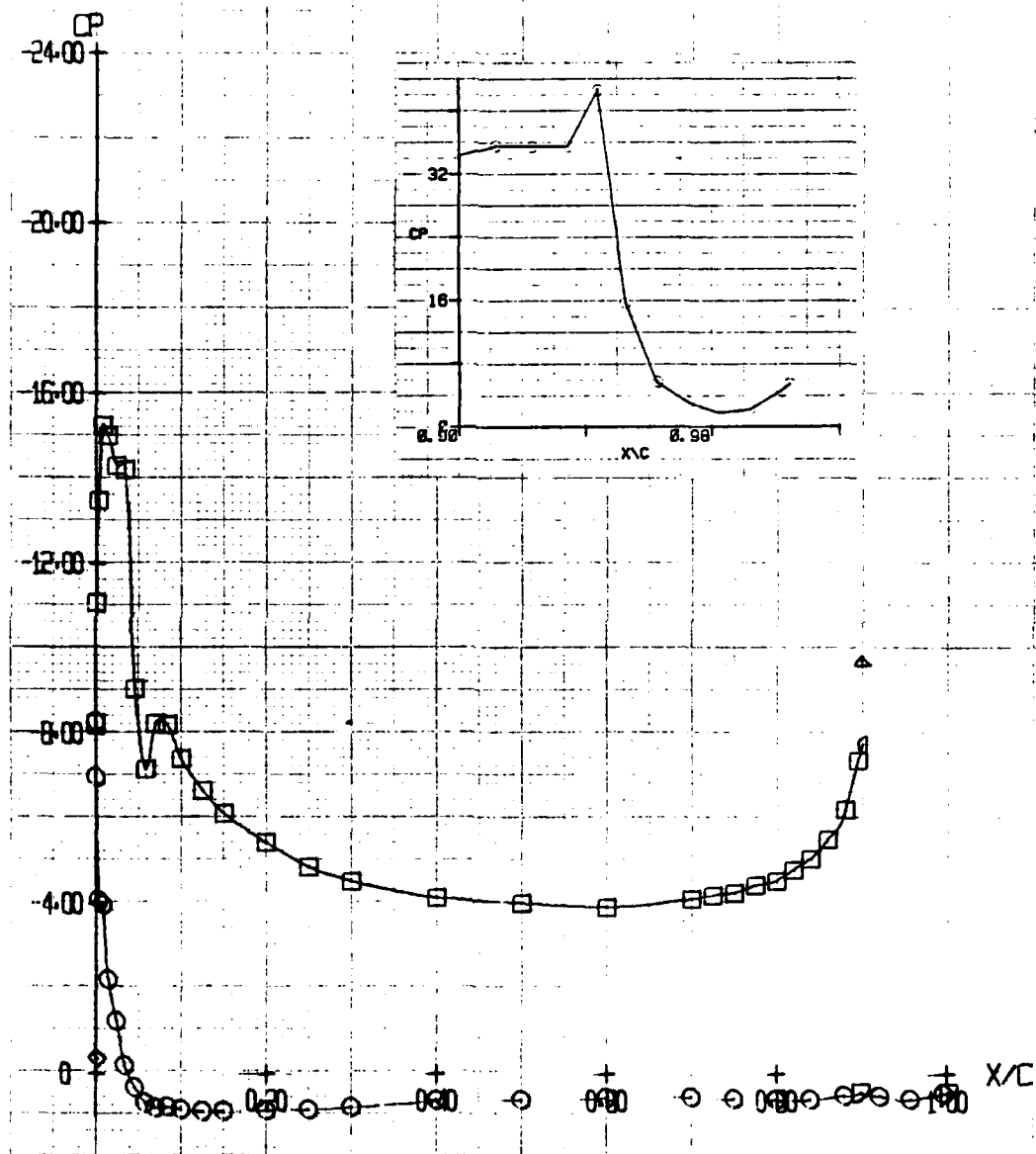


Figure I-17. $C_\mu = 2.04$, $\Theta = 50^\circ$, $\alpha = 0^\circ$ (Test 272, Run 21)
 $Q = 7.04$ psf

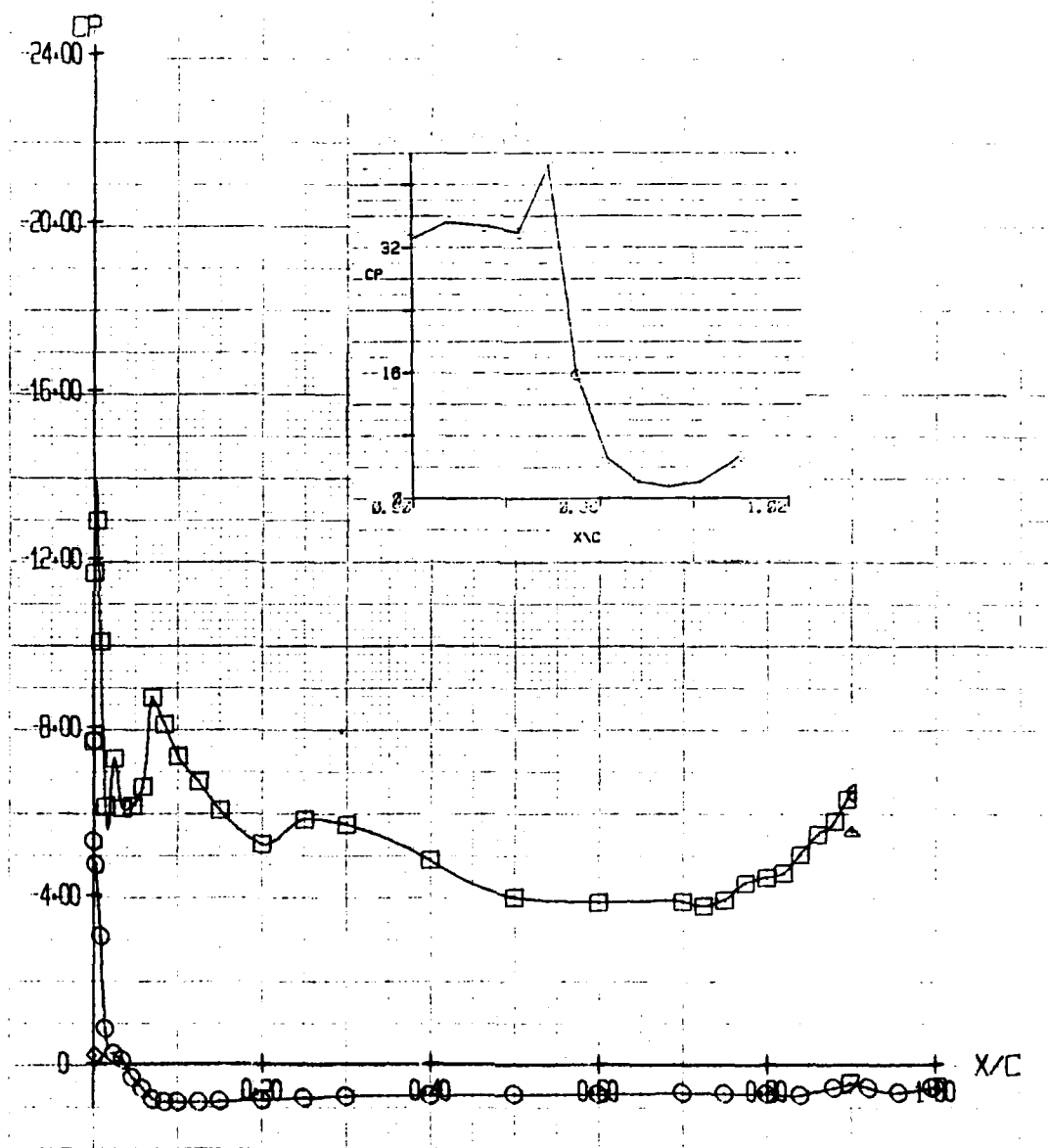


Figure I-18. $C_{\mu} = 2.04$, $\Theta = 50^\circ$, $\alpha = 0^\circ$ (Test 272, Run 29)
 $Q = 7.03$ psf

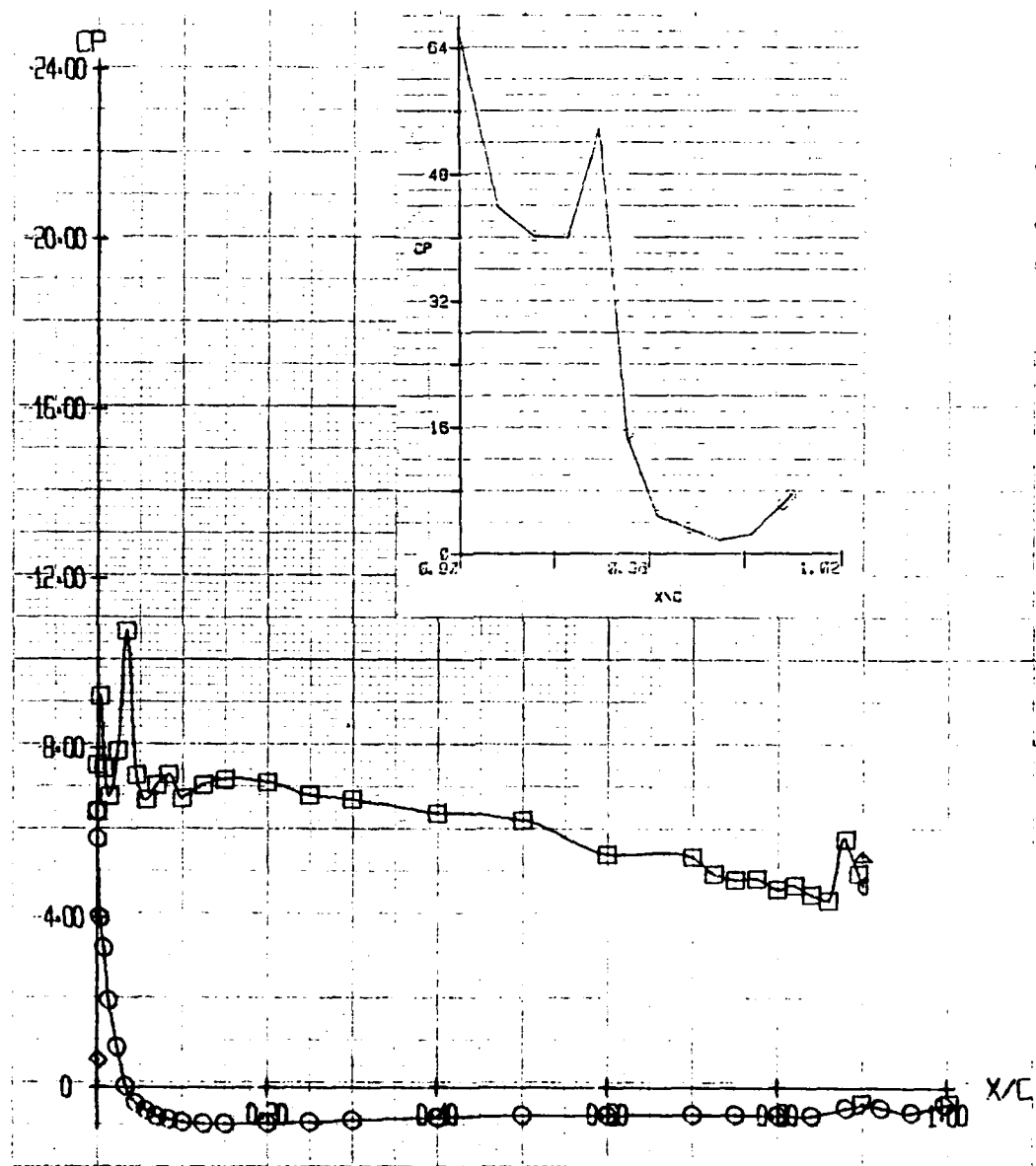


Figure I-19. $C_{\mu} = 3.06$, $\Theta = 50^\circ$, $\alpha = 0^\circ$ (Test 272, Run 28)
 $Q = 7.06$ psf

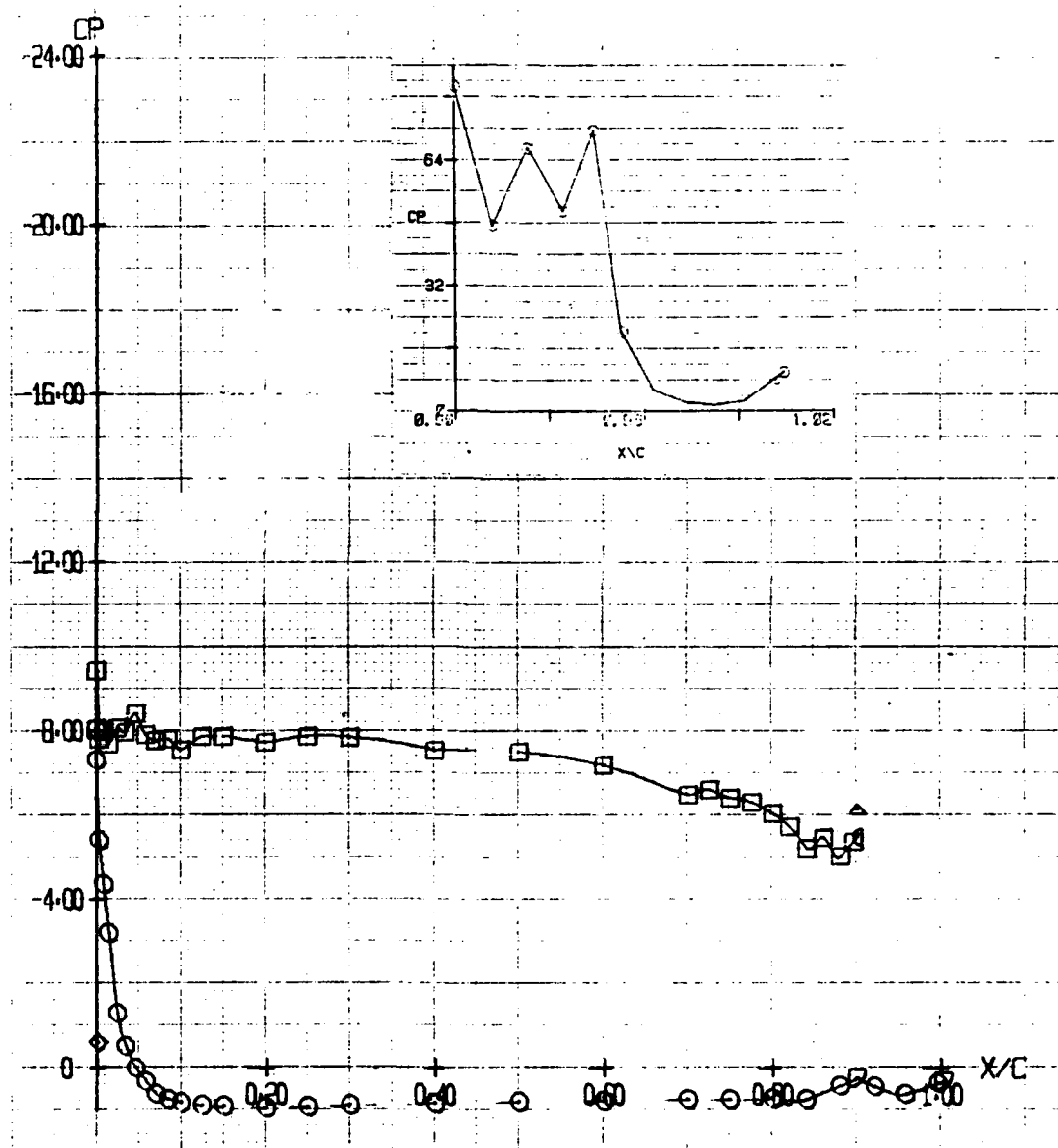


Figure I-20. $C_{\mu} = 4.15$, $\Theta = 50^\circ$, $\alpha = 0^\circ$ (Test 272, Run 22)
 $Q = 6.00$ psf

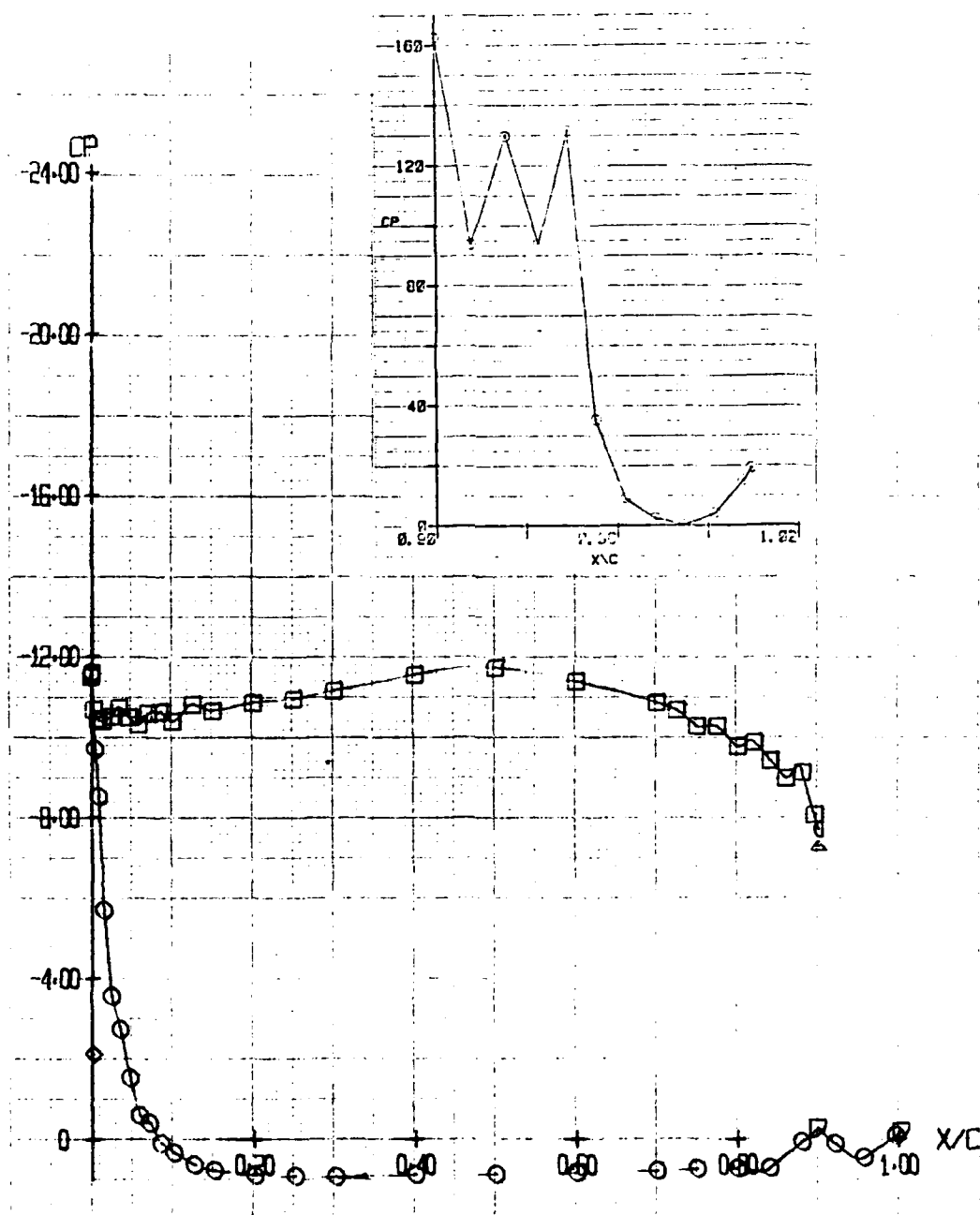


Figure I-21. $C_{\mu} = 8.17$, $\Theta = 50^\circ$, $\alpha = 0^\circ$ (Test 272, Run 25)
 $Q = 3.02$ psf

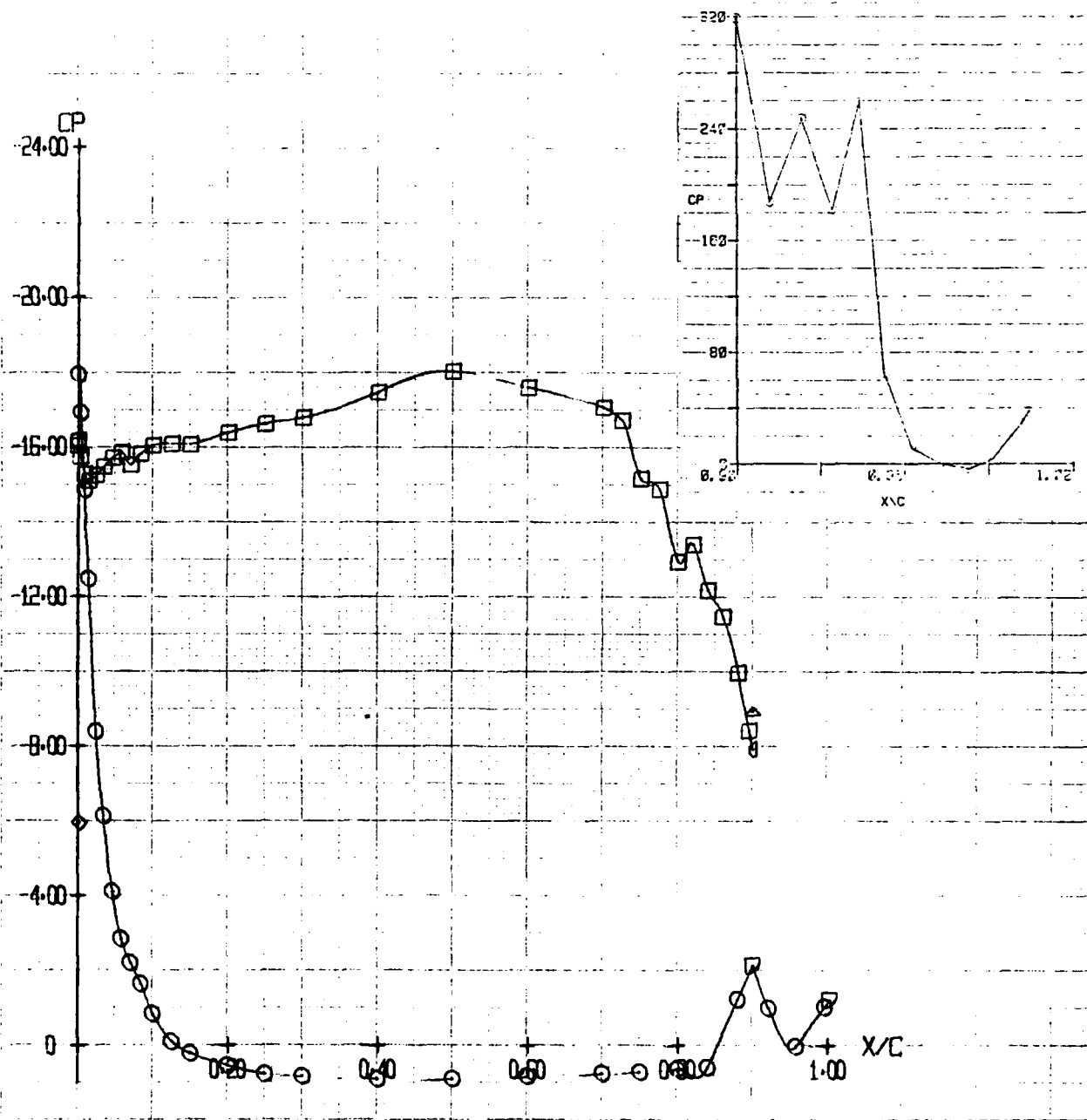


Figure I-22. $C_\mu = 16.22$, $\Theta = 50^\circ$, $\alpha = 0^\circ$ (Test 272, Run 26)
 $Q = 1.51$ psf

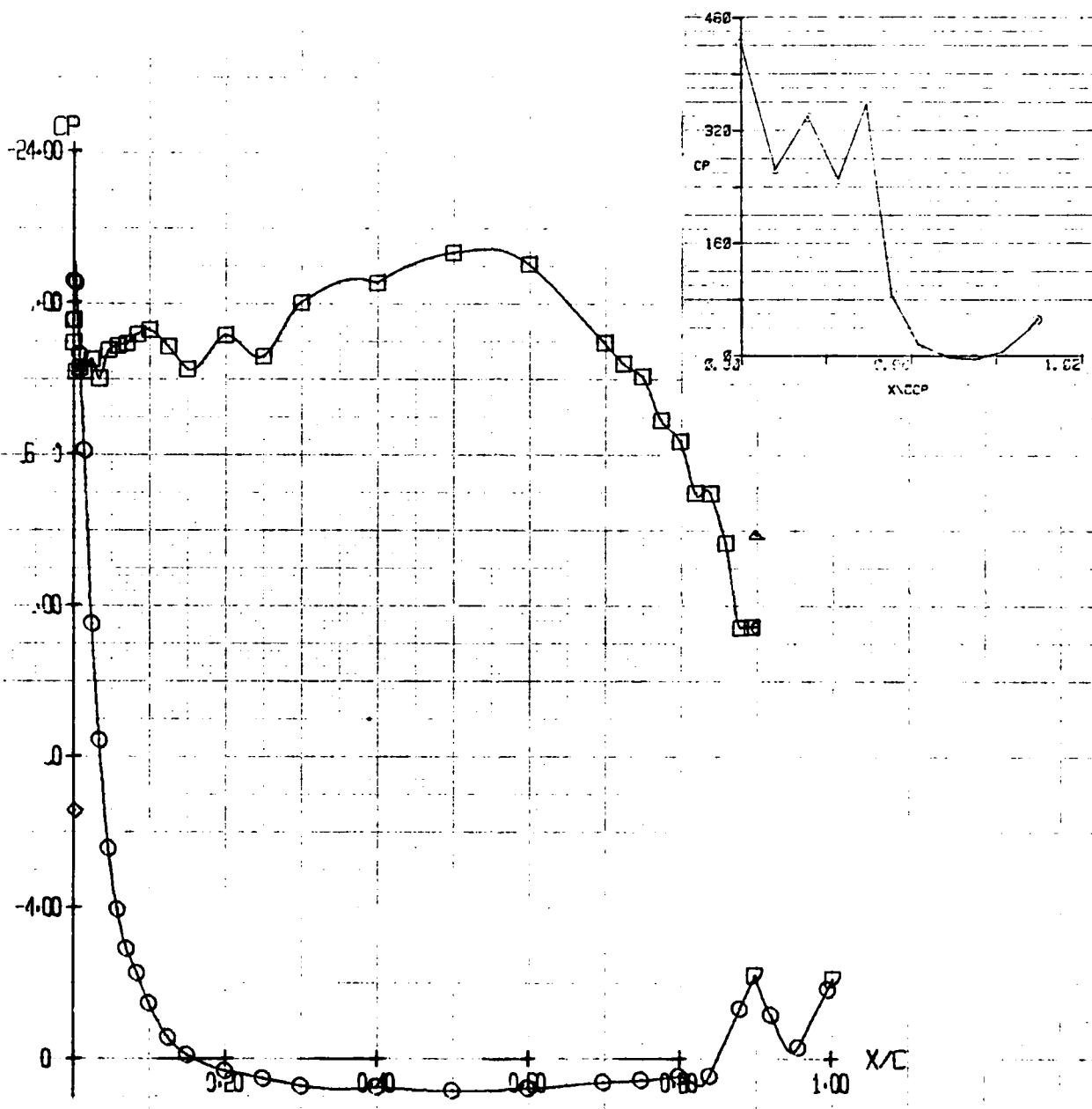


Figure I-23. $C_u = 22.33$, $\Theta = 50^\circ$, $\alpha = 0^\circ$ (Test 272, Run 27)
 $Q = 1.09$ psf

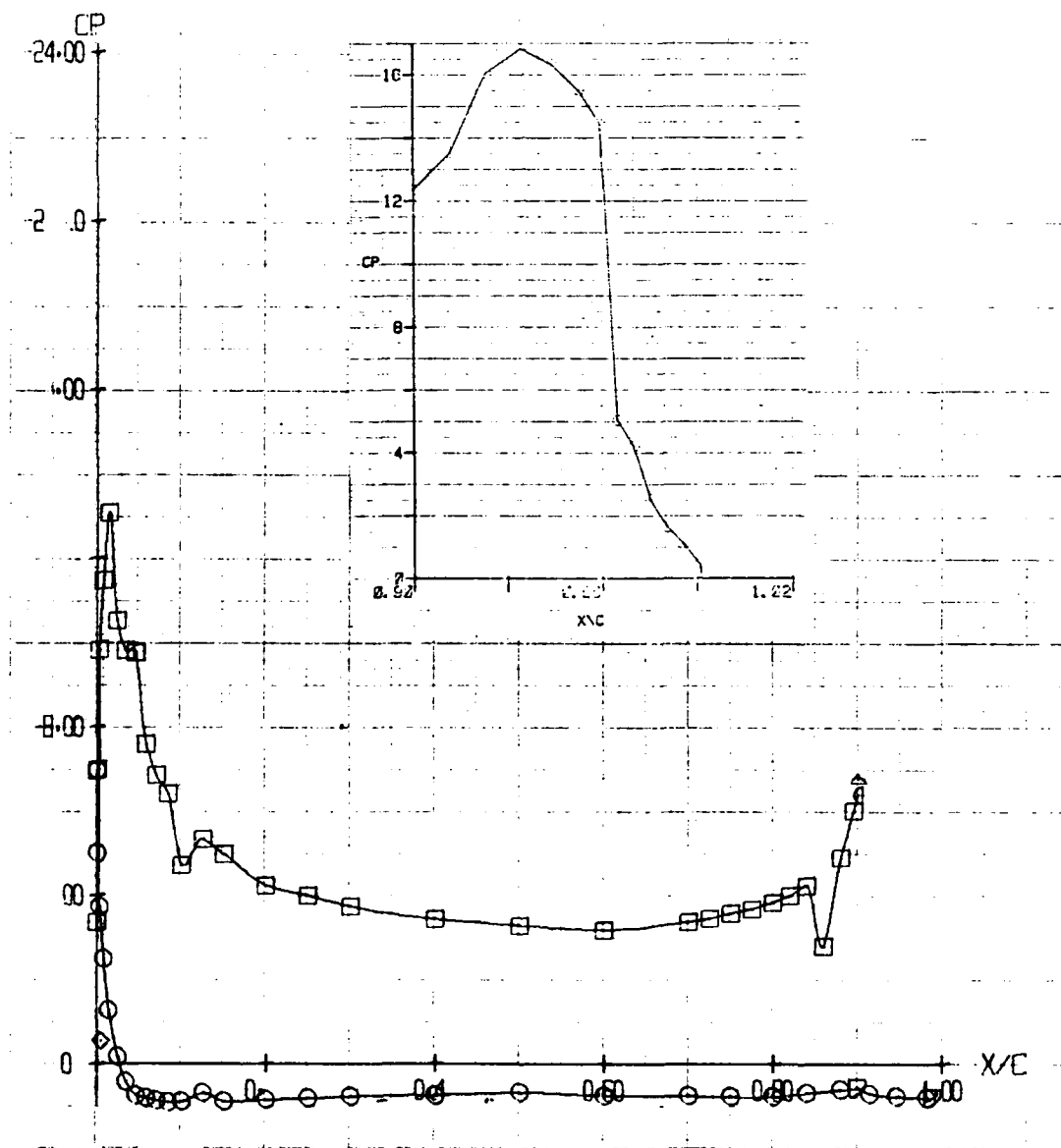


Figure I-24. $C_u = 0.55$, $\Theta = 70^\circ$, $\alpha = 0^\circ$ (Test 272, Run 30)
 $Q = 6.84$ psf

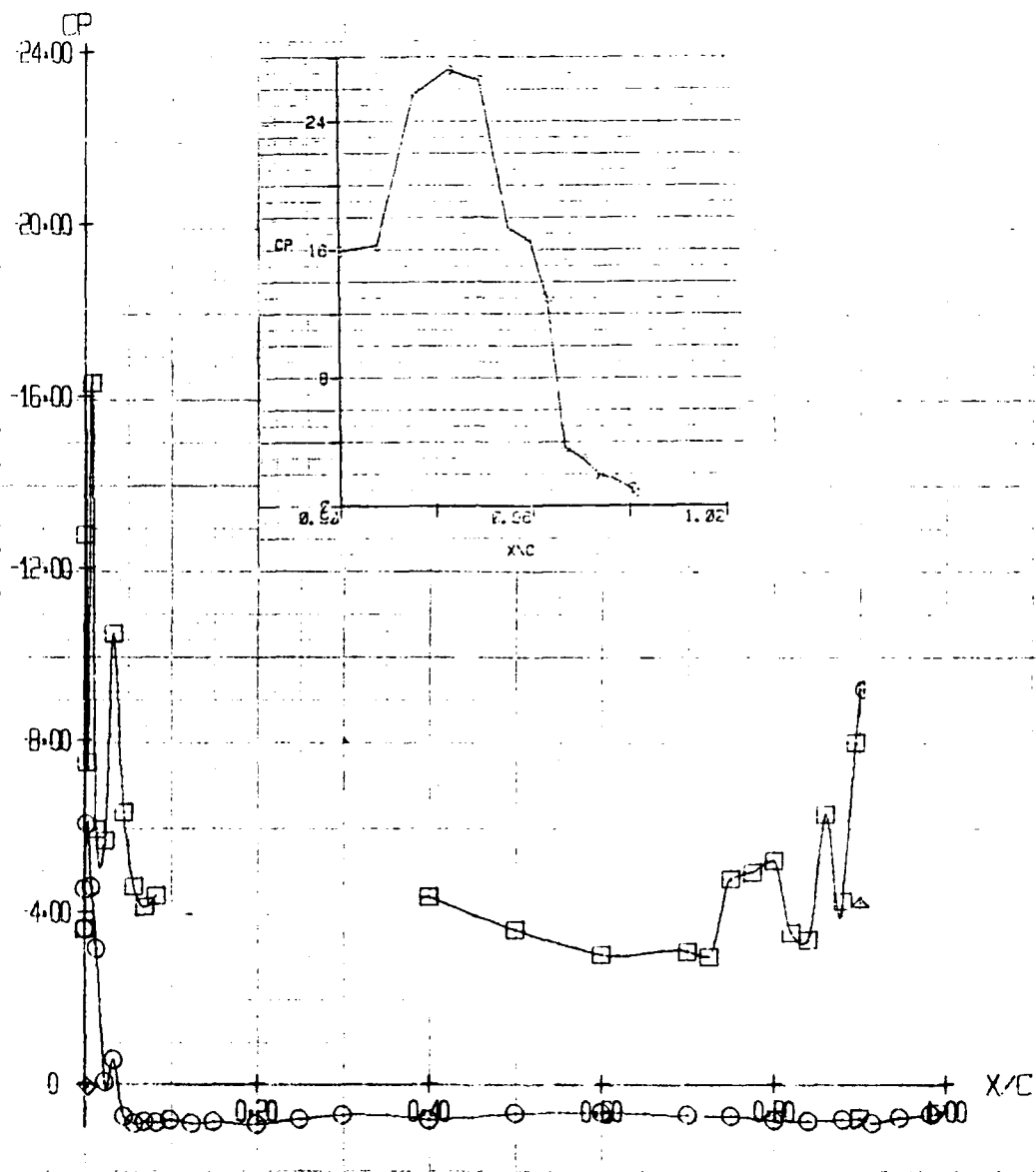


Figure I-25. $C_u = 1.04$, $\Theta = 70^\circ$, $\alpha = 0^\circ$ (Test 272, Run 43)
 $Q = 6.93$ psf

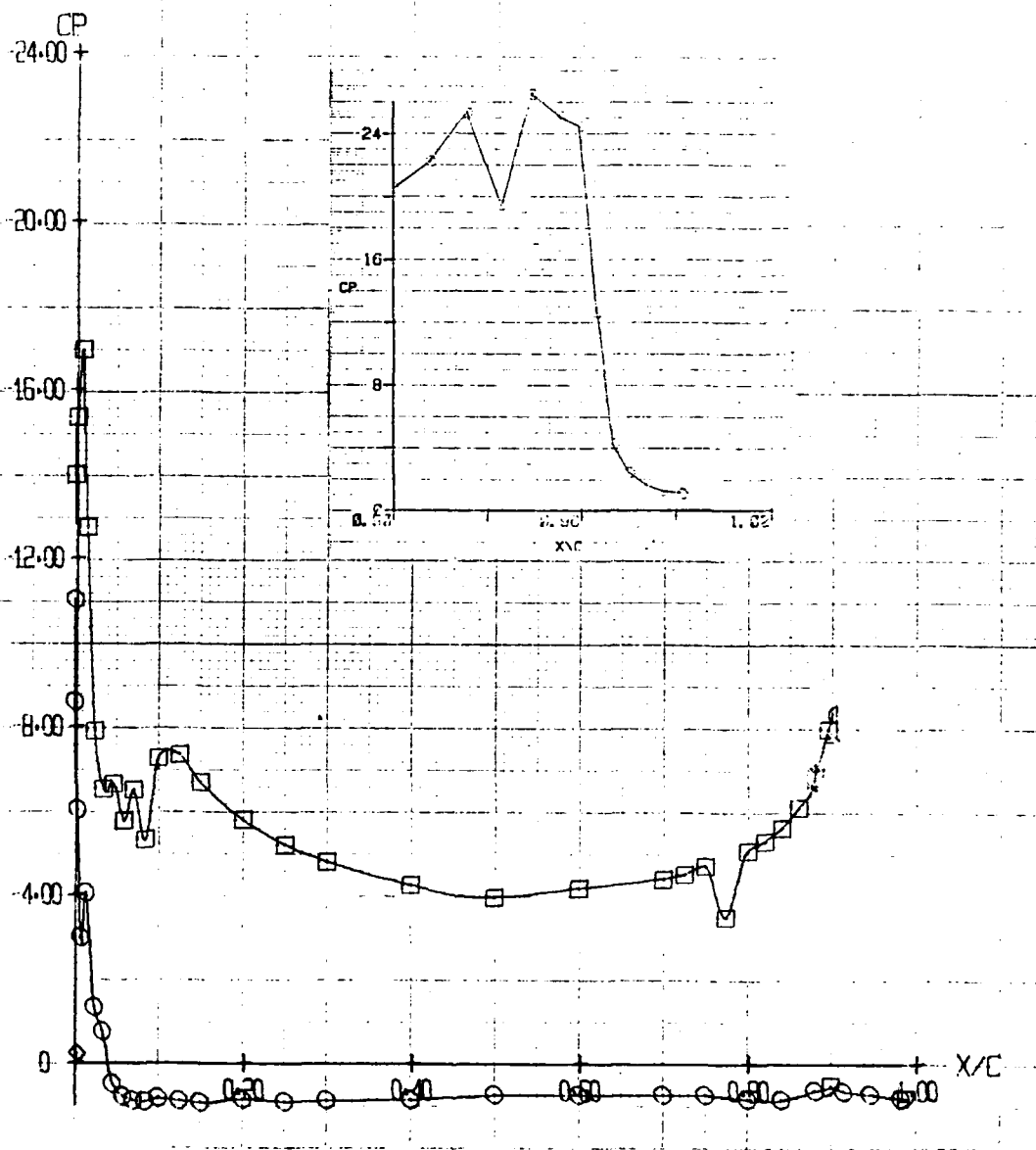


Figure I-26. $C_u = 1.05$, $\Theta = 70^\circ$, $\alpha = 0^\circ$ (Test 272, Run 31)
 $Q = 6.94$ psf

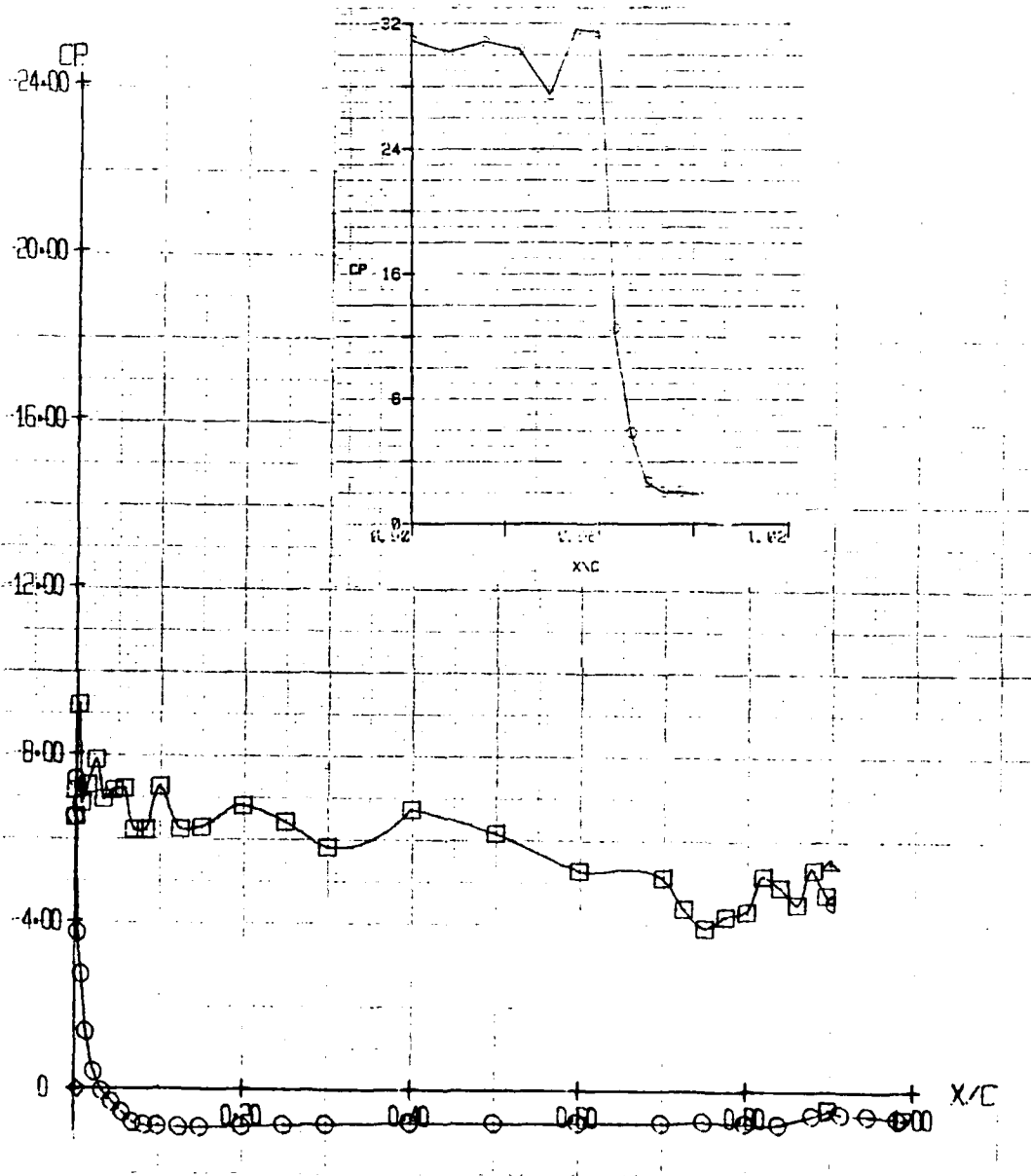


Figure I-27. $C_u = 2.08$, $\Theta = 70^\circ$, $\alpha = 0^\circ$ (Test 272, Run 32)
 $Q = 7.02$ psf

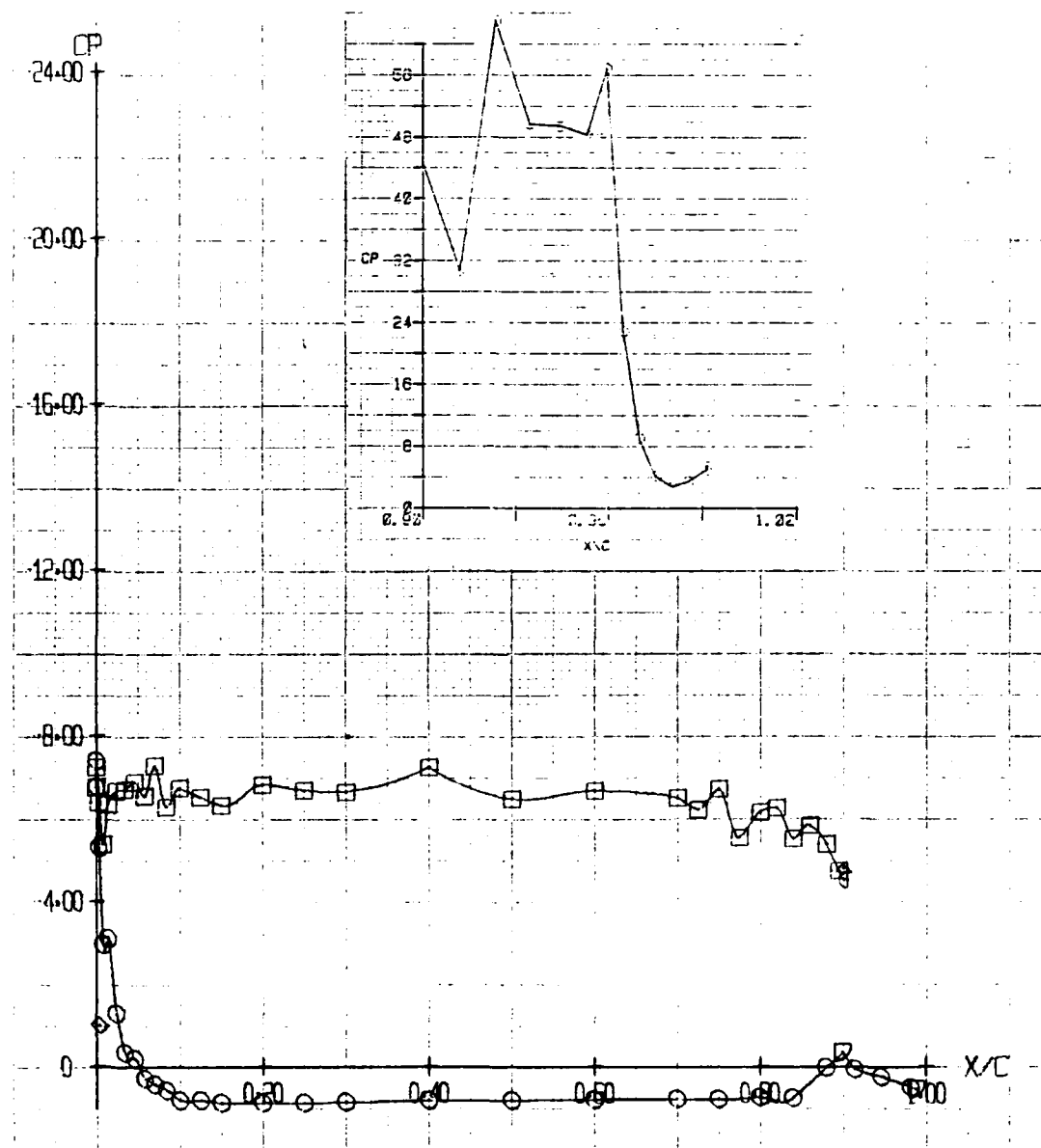


Figure I-28. $C_u = 4.26$, $\Theta = 70^\circ$, $\alpha = 0^\circ$ (Test 272, Run 36)
 $Q = 5.99$ psf

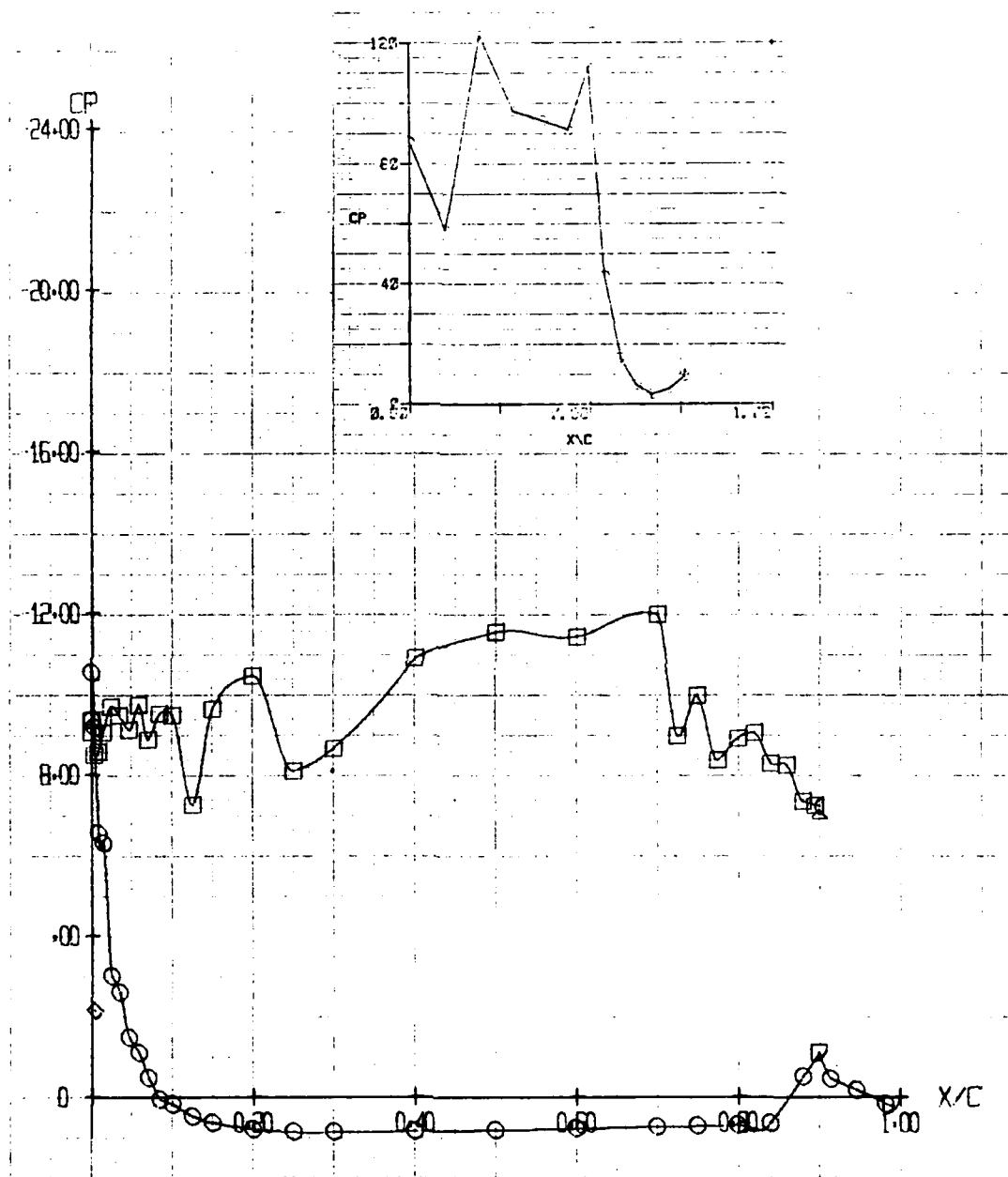


Figure I-29. $C_{\mu} = 8.31$, $\Theta = 70^\circ$, $\alpha = 0^\circ$ (Test 272, Run 34)
 $Q = 3.06$ psf

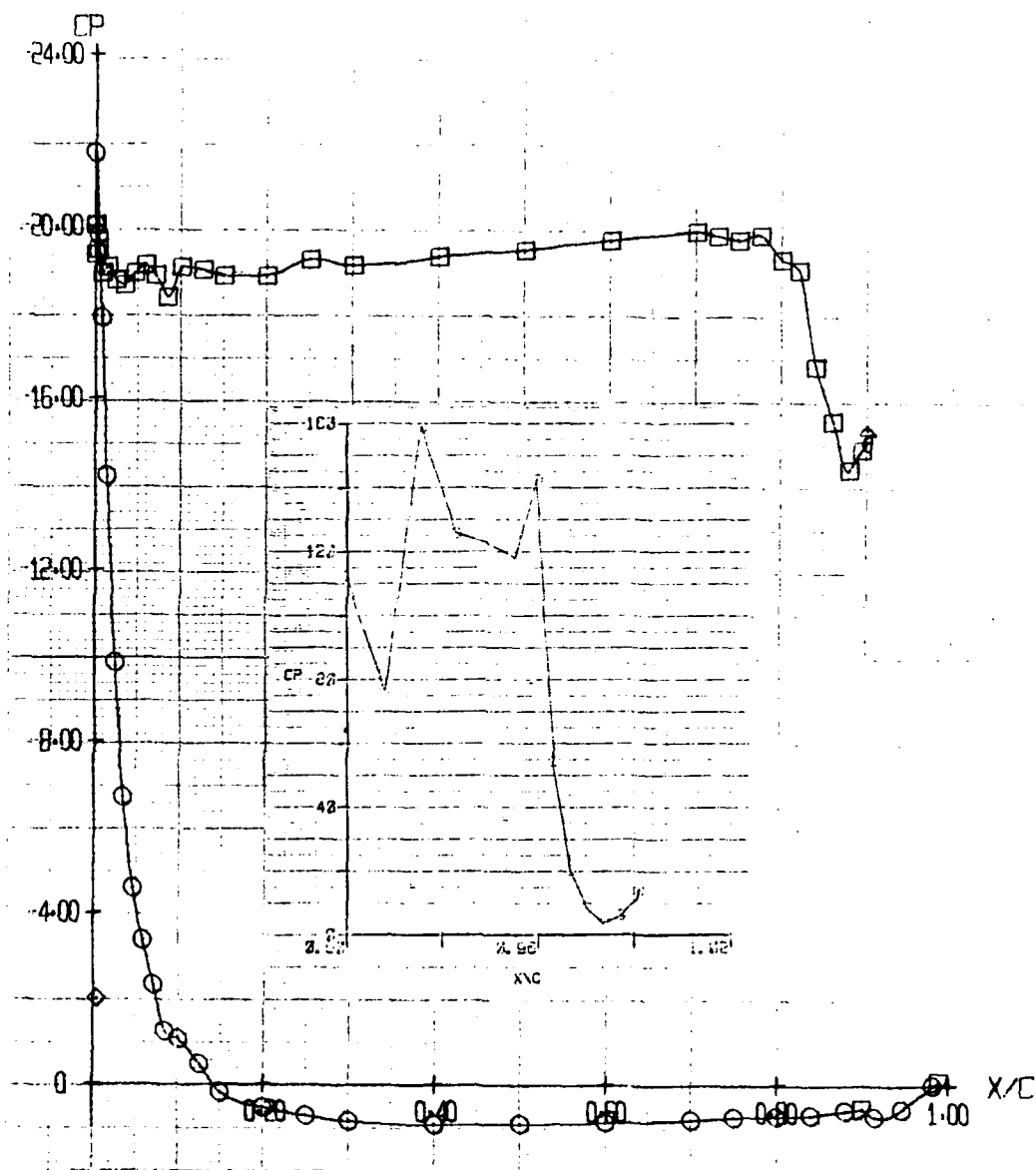


Figure I-30. $C_{\mu} = 10.52$, $\Theta = 70^\circ$, $\alpha = 0^\circ$ (Test 272, Run 42)
 $Q = 2.38$ psf

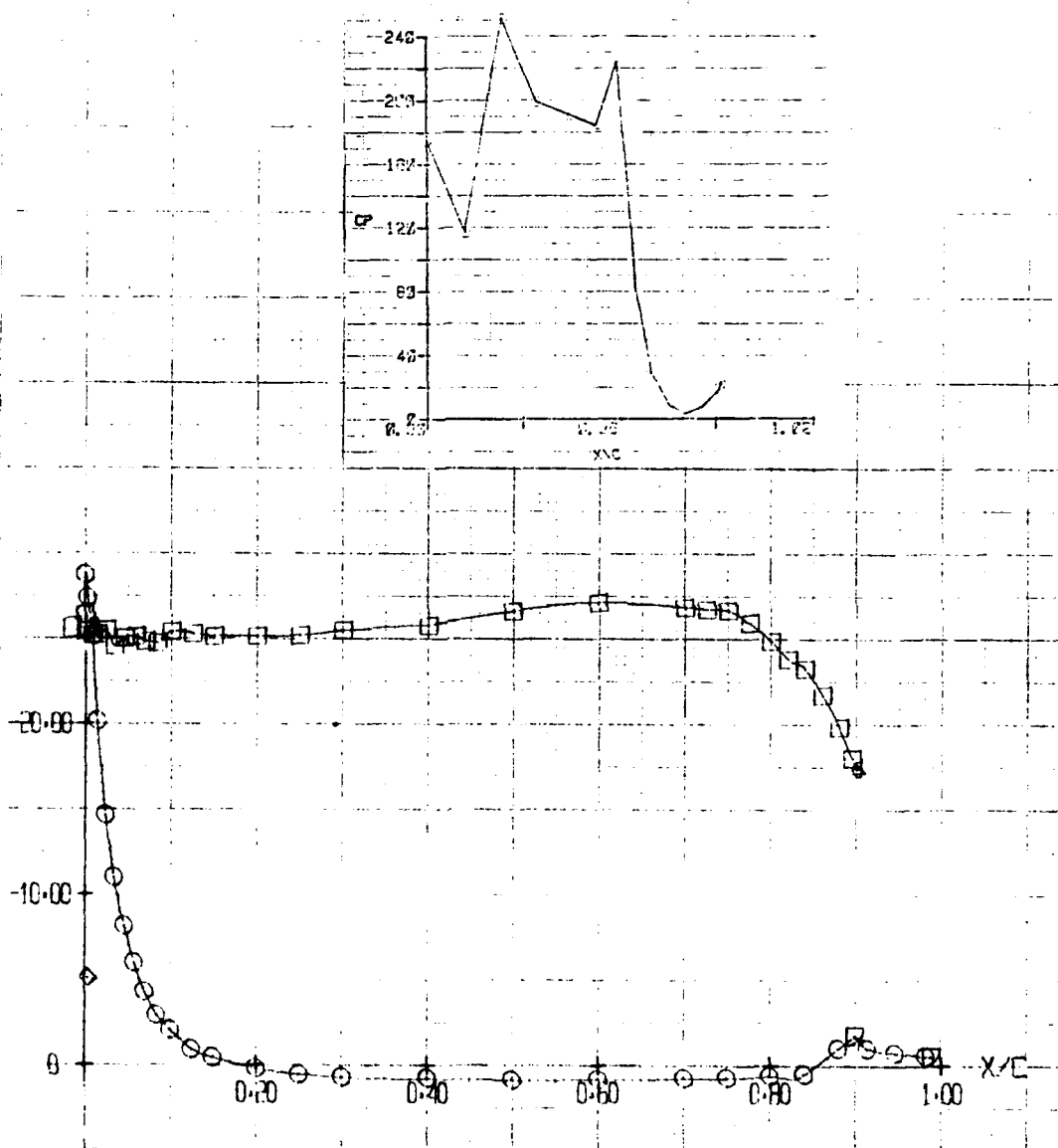


Figure I-31. $C_u = 16.60$, $\Theta = 70^\circ$, $\alpha = 0^\circ$ (Test 272, Run 37)
 $Q = 1.51$ psf

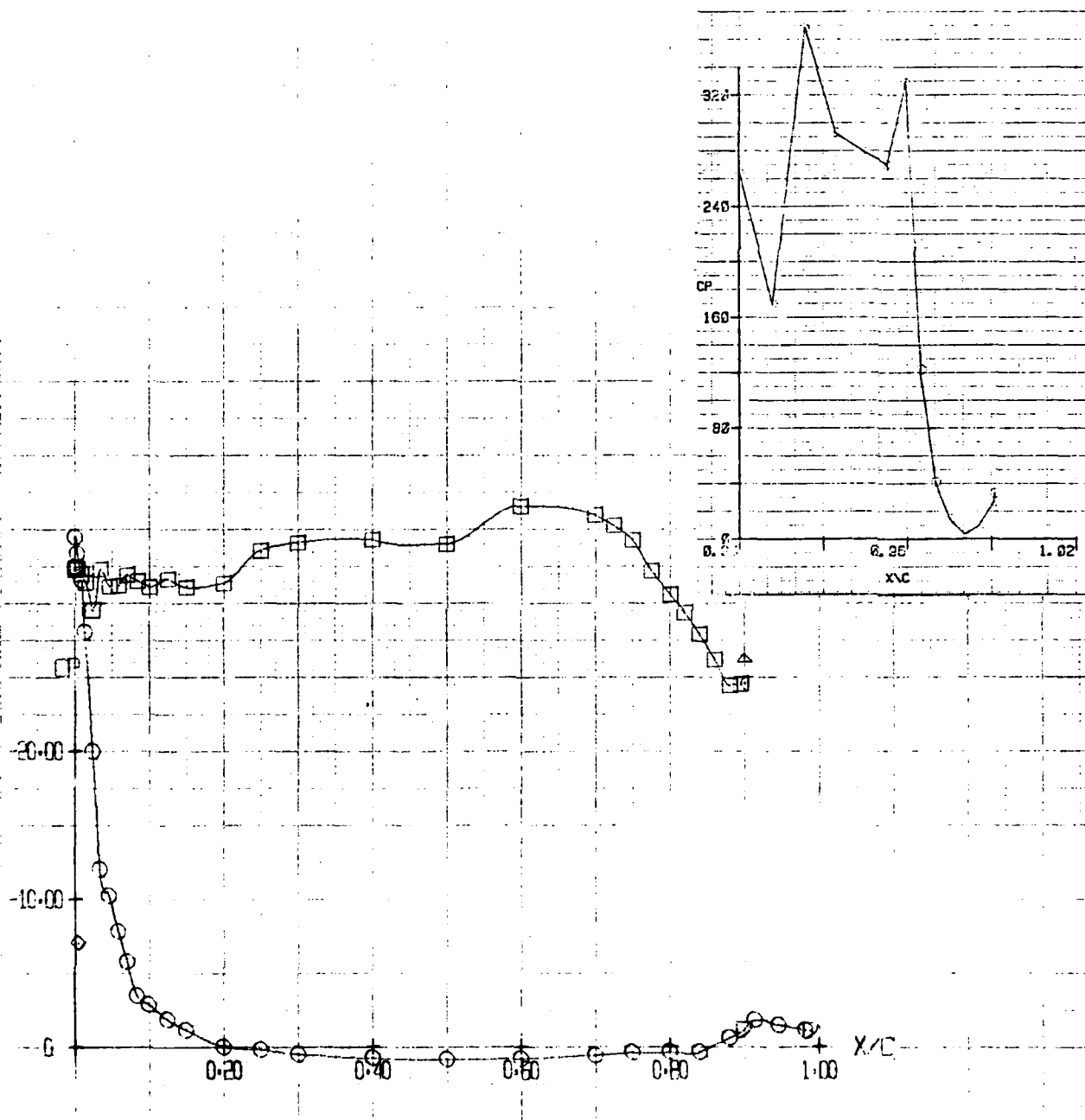


Figure I-32. $C_{\mu} = 24.43$, $\Theta = 70^\circ$, $\alpha = 0^\circ$ (Test 272, Run 38)
 $Q = 1.02$ psf

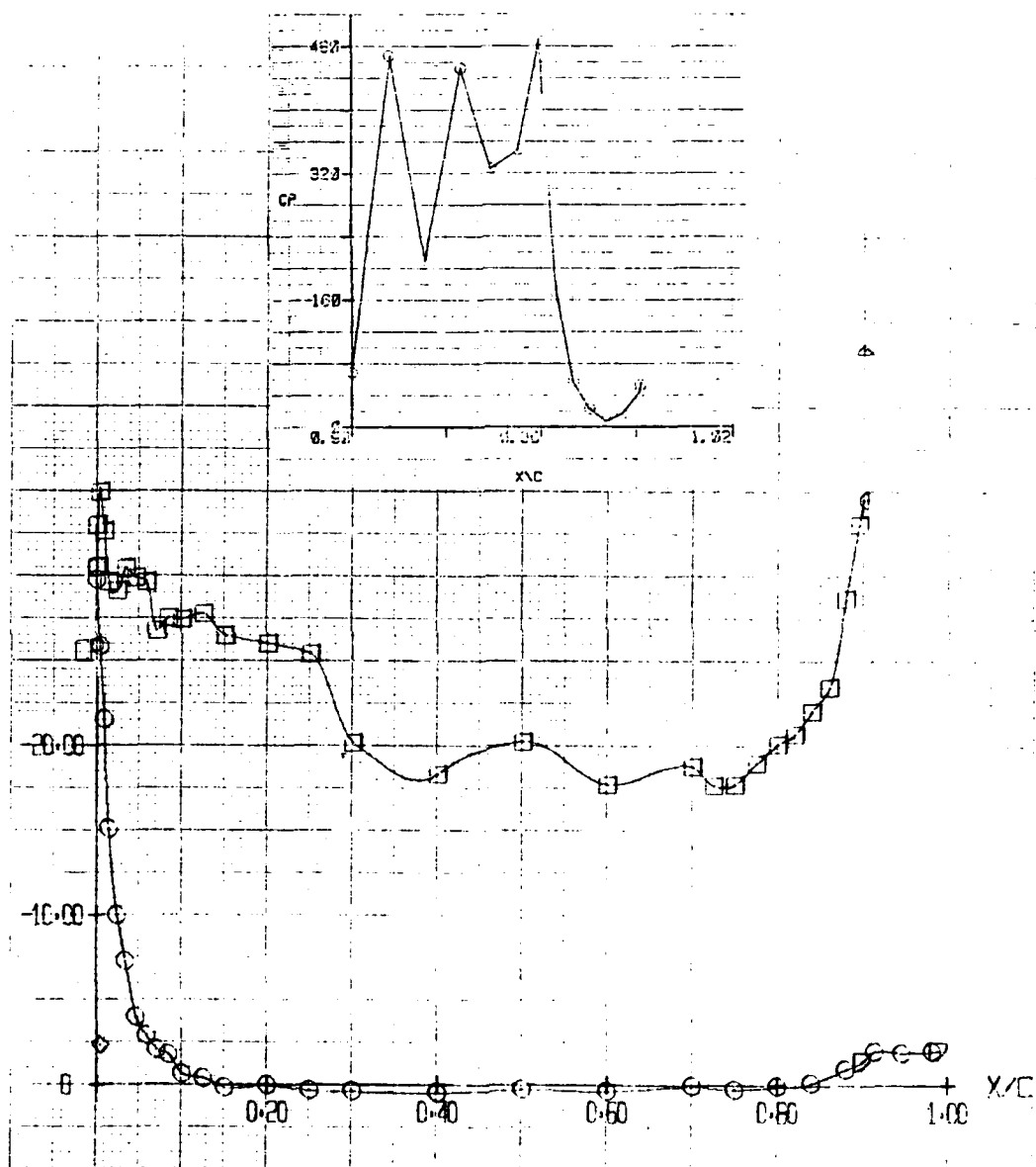


Figure I-33. $C_u = 28.92$, $\Theta = 70^\circ$, $\alpha = 0^\circ$ (Test 272, Run 39)
 $Q = 1.05$ psf

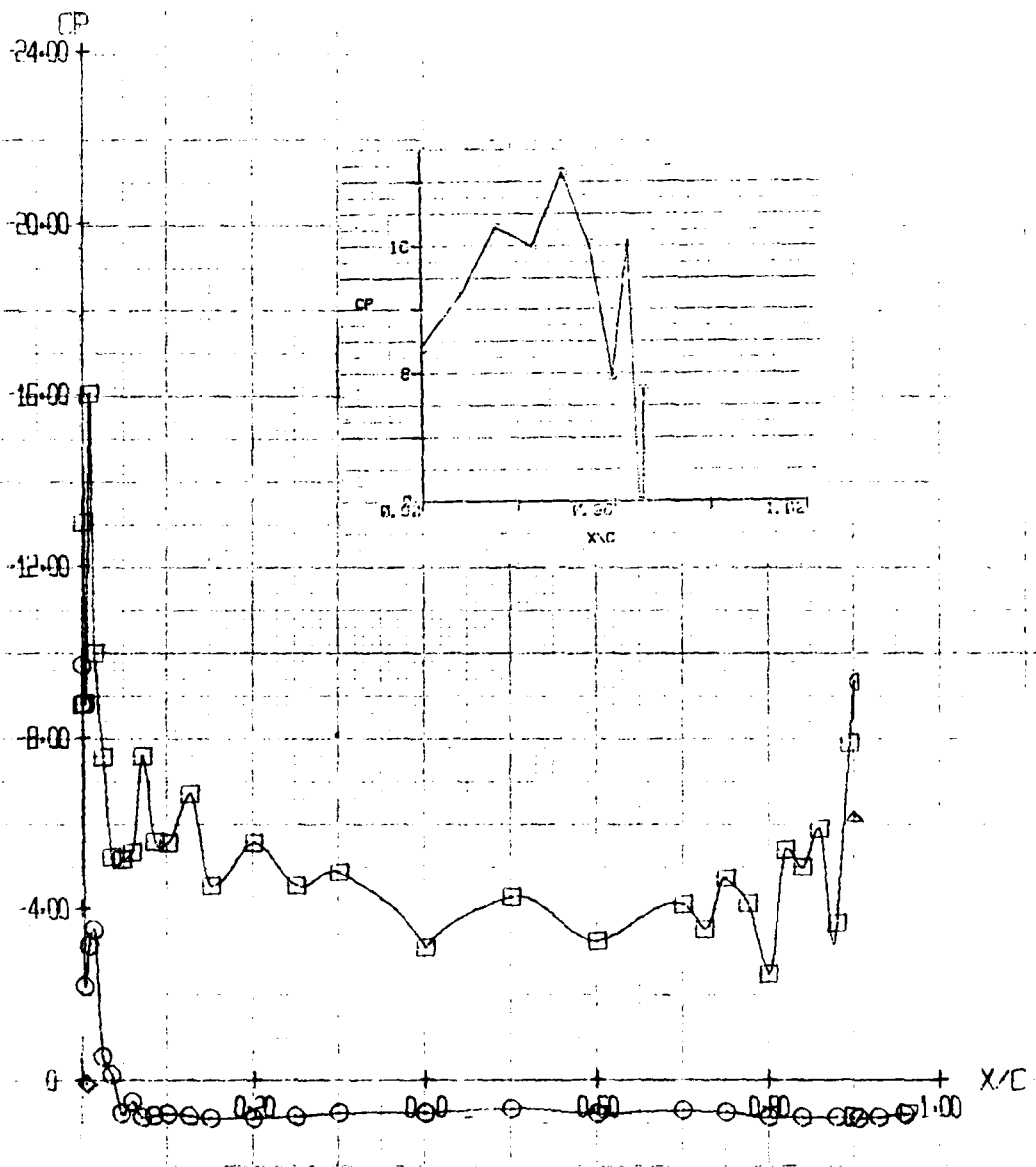


Figure I-34. $C_u = 0.52$, $\Theta = 90^\circ$, $\alpha = 0^\circ$ (Test 272, Run 47)
 $Q = 7.02$ psf

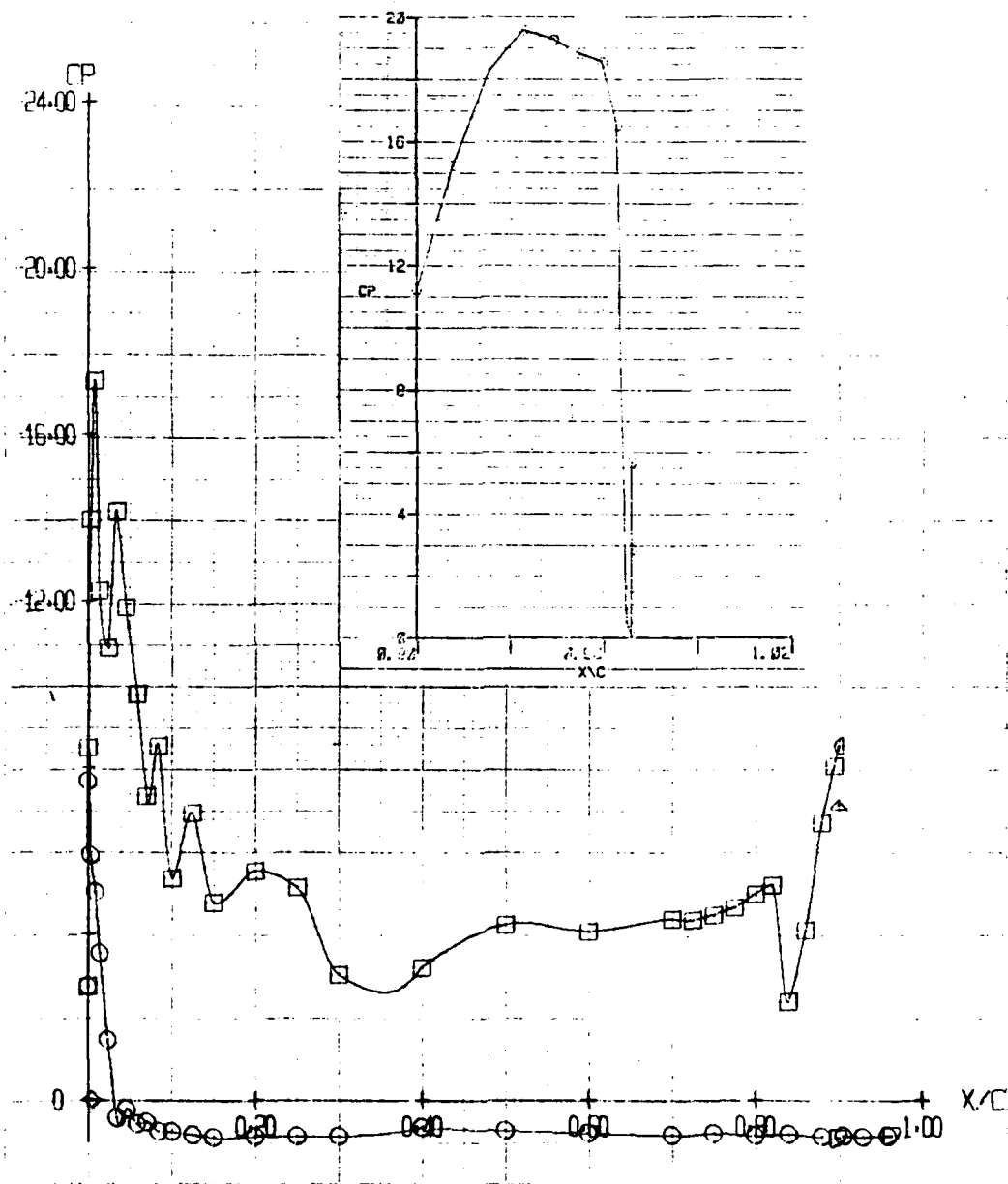


Figure I-35. $C_{\mu} = 0.51$, $\Theta = 90^\circ$, $\alpha = 0^\circ$ (Test 272, Run 44)
 $Q = 7.24$ psf

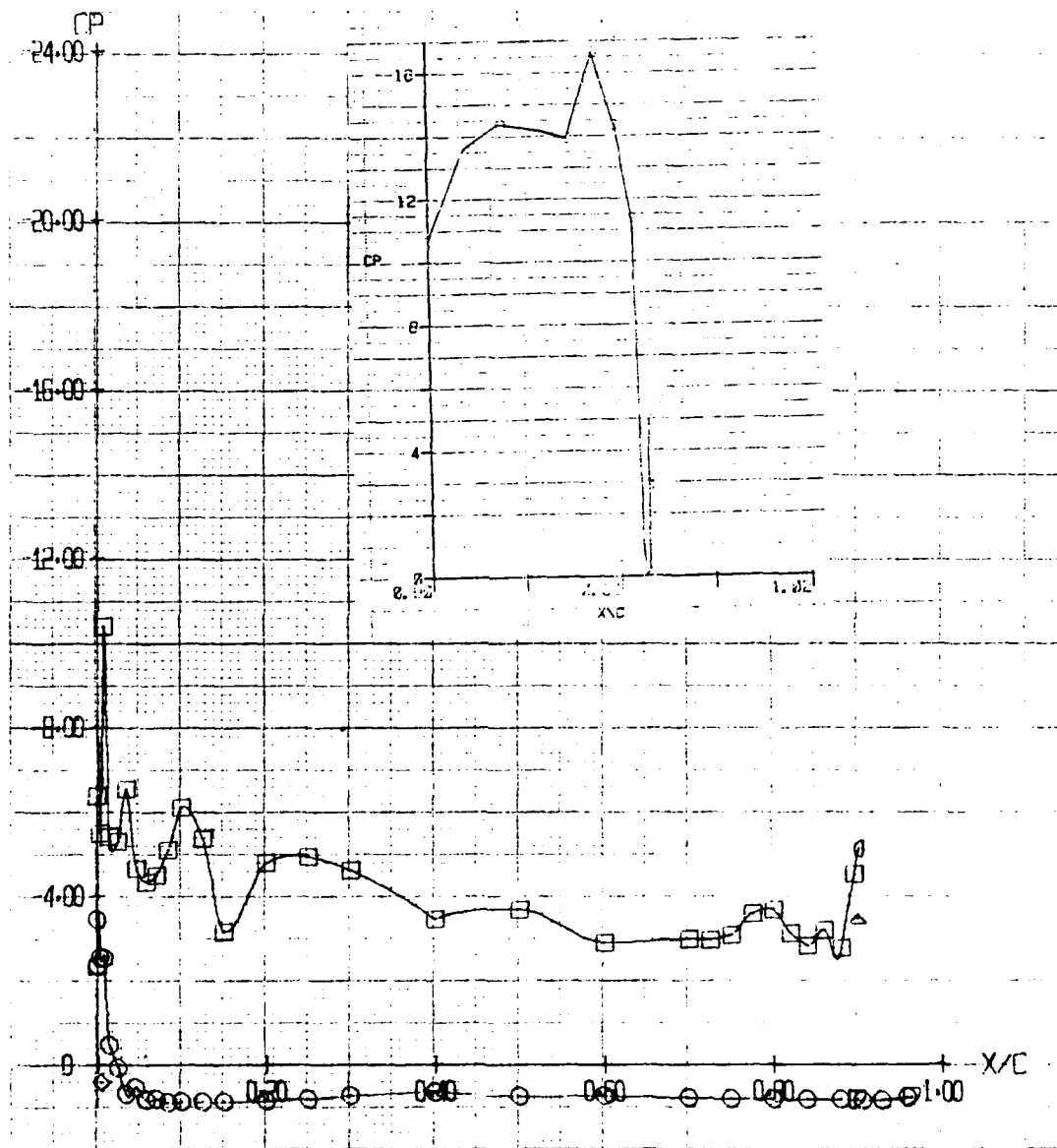


Figure I-36. $C_u = 0.97$, $\theta = 90^\circ$, $\alpha = 0^\circ$ (Test 272, Run 45)
 $Q = 6.95$ psf

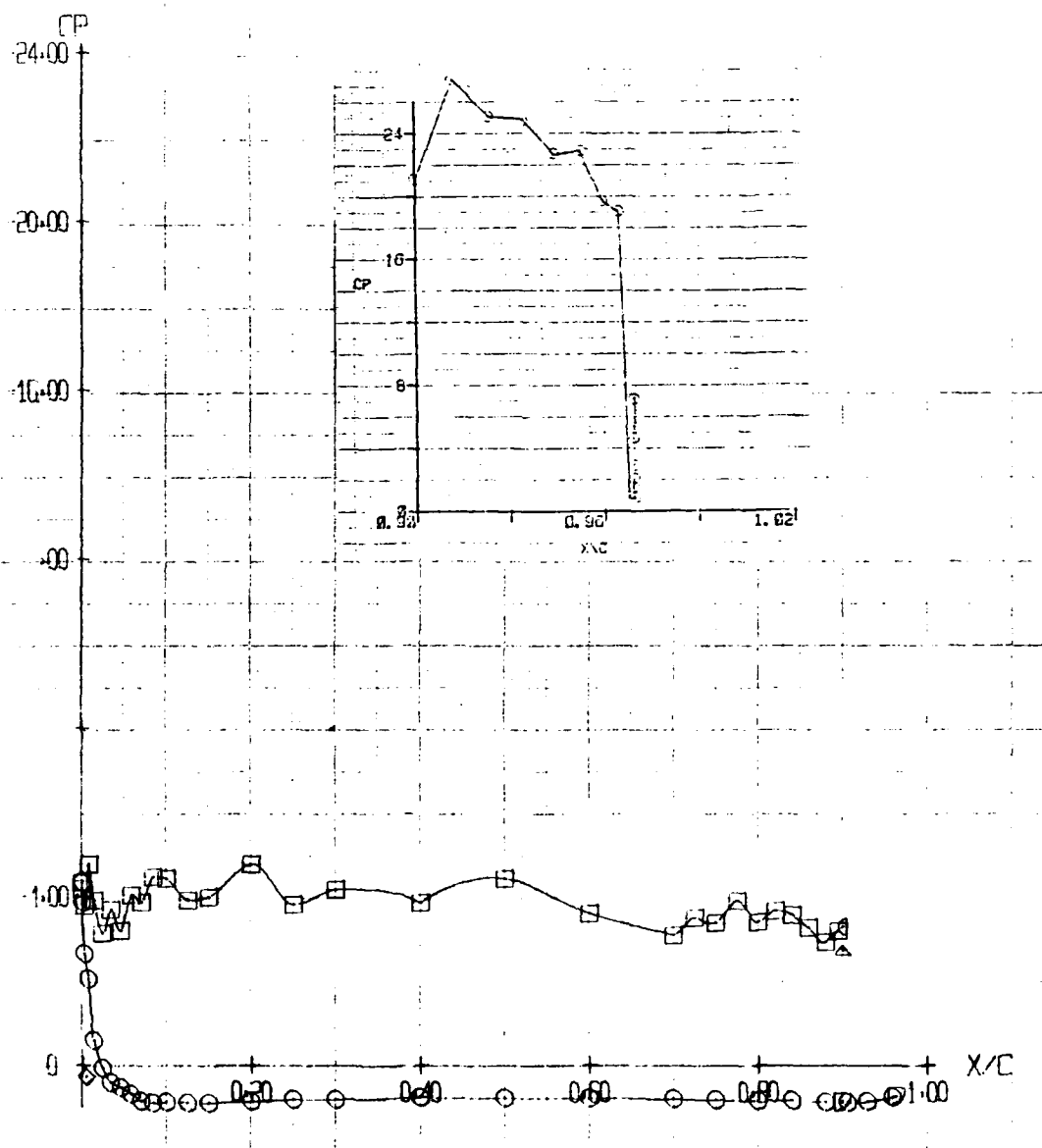


Figure I-37. $C_{\mu} = 1.95$, $\Theta = 90^\circ$, $\alpha = 0^\circ$ (Test 272, Run 56)
 $Q = 6.98$ psf

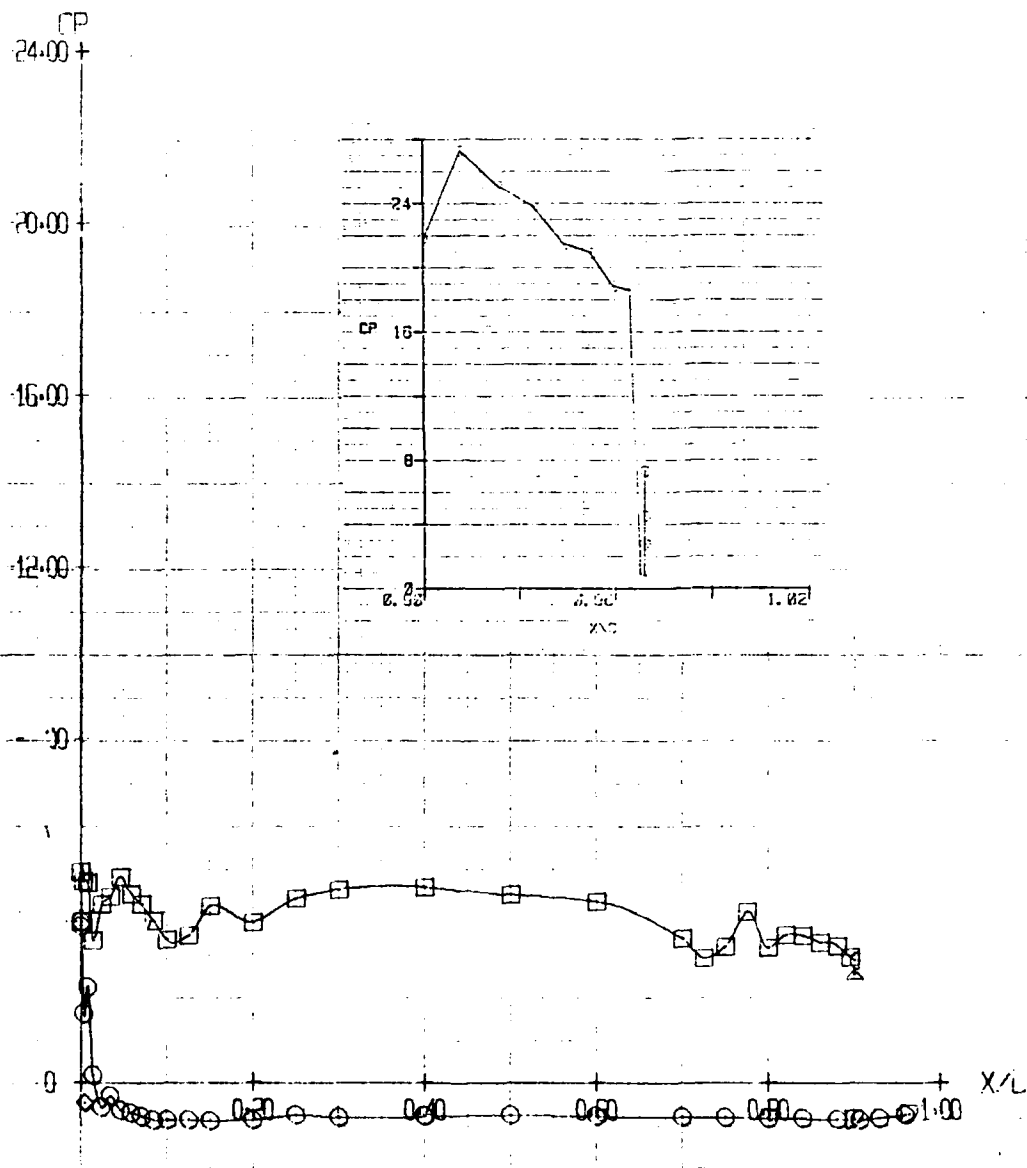


Figure I-38. $C_u = 1.96$, $\Theta = 90^\circ$, $\alpha = 0^\circ$ (Test 272, Run 46)
 $Q = 6.99$ psf

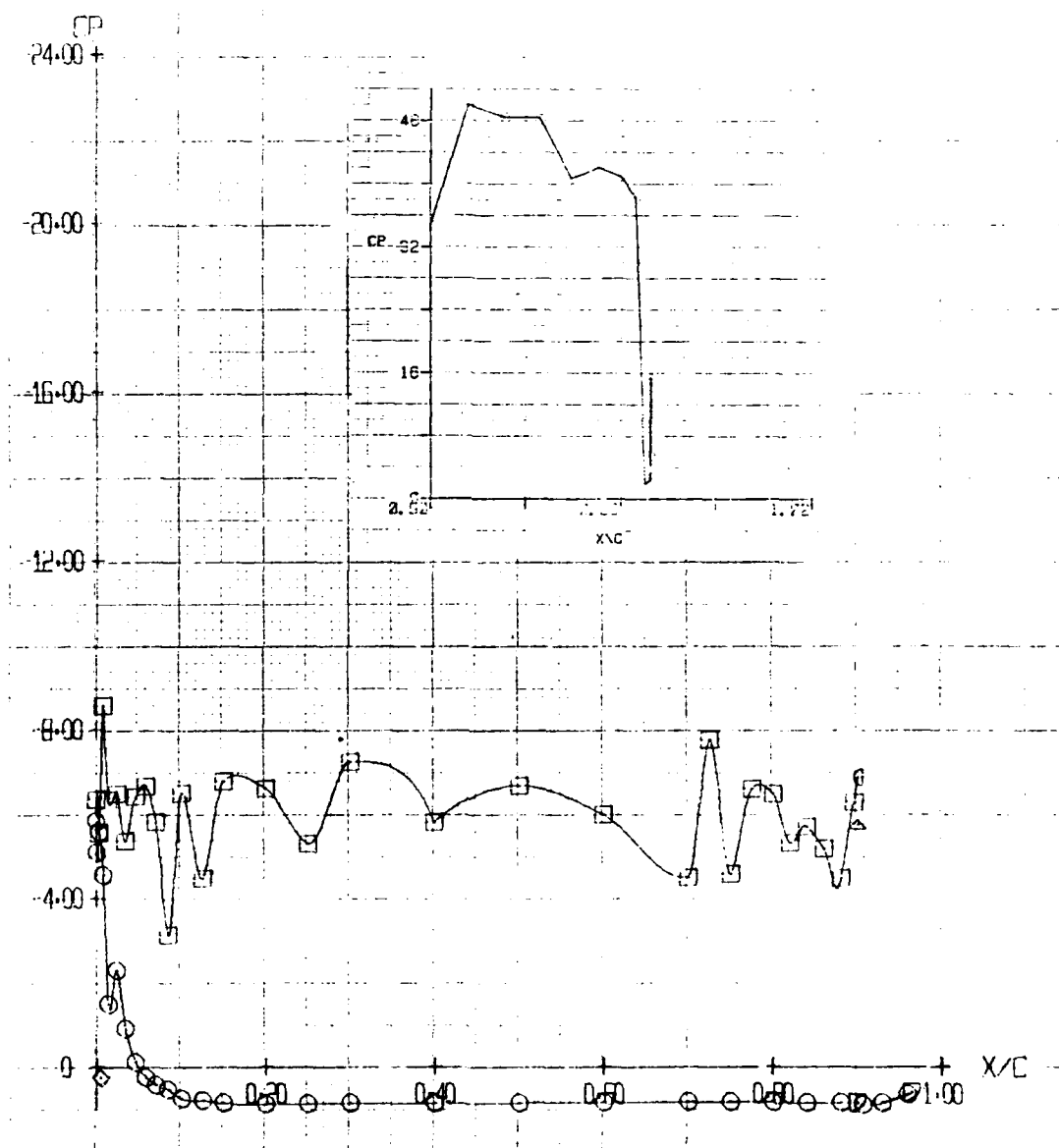


Figure I-39. $C_{\mu} = 3.93$, $\Theta = 90^\circ$, $\alpha = 0^\circ$ (Test 272, Run 48)
 $Q = 5.99$ psf

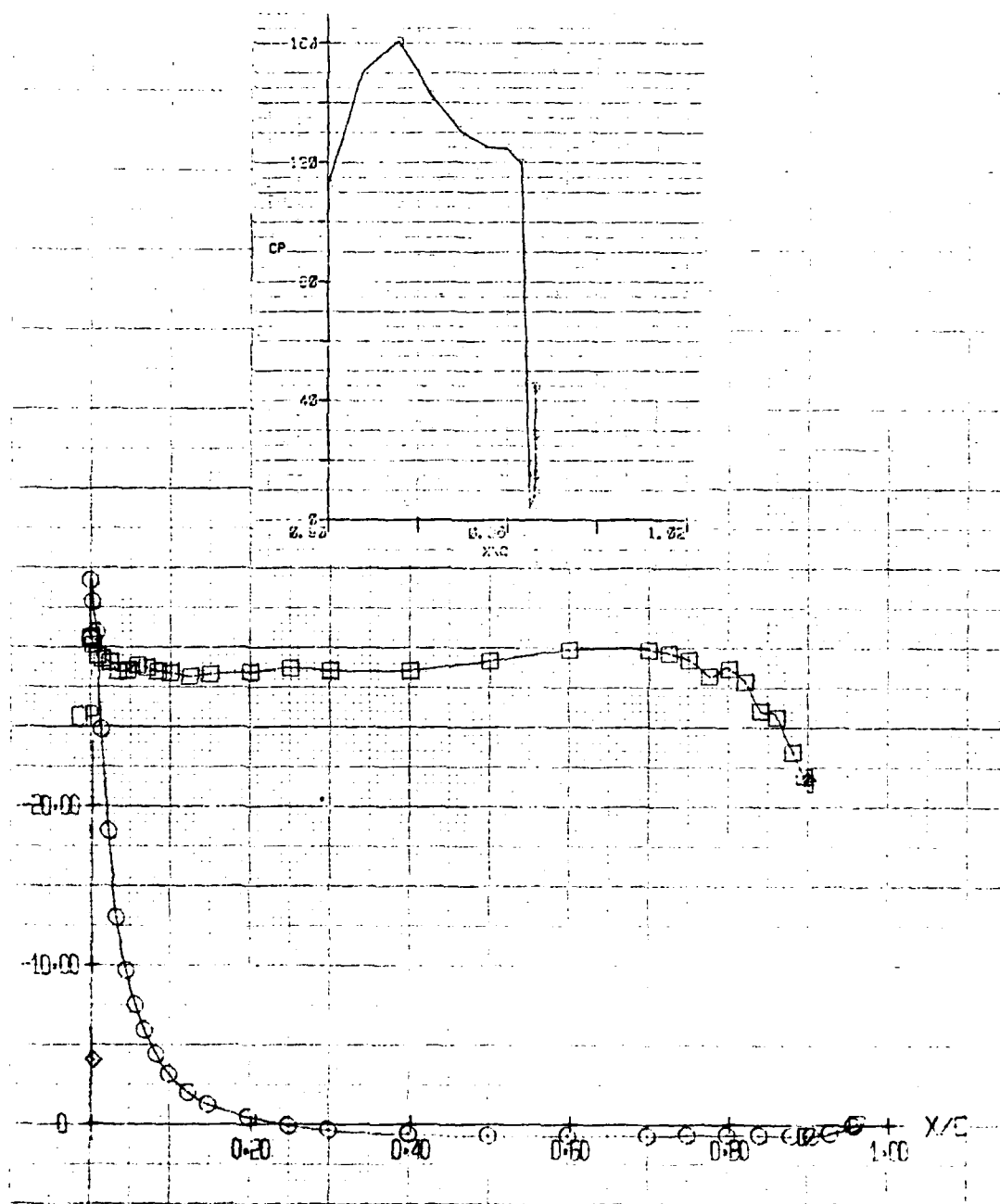


Figure I-40. $C_{\mu} = 11.49$, $\Theta = 90^\circ$, $\alpha = 0^\circ$ (Test 272, Run 49)
 $Q = 2.02$ psf

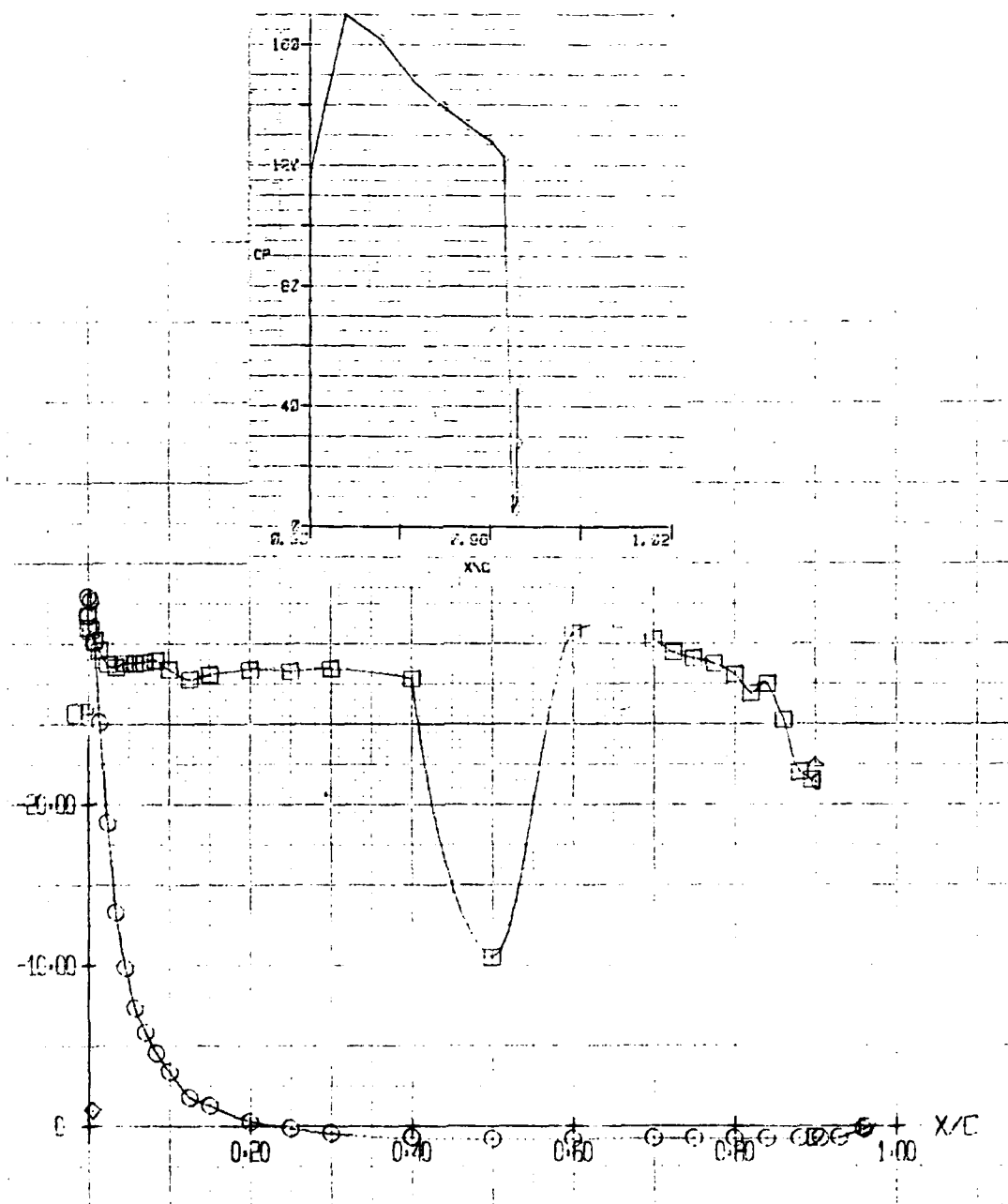


Figure I-41. $C_\mu = 11.80$, $\Theta = 90^\circ$, $\alpha = 0^\circ$ (Test 272, Run 54)
 $Q = 1.97$ psf

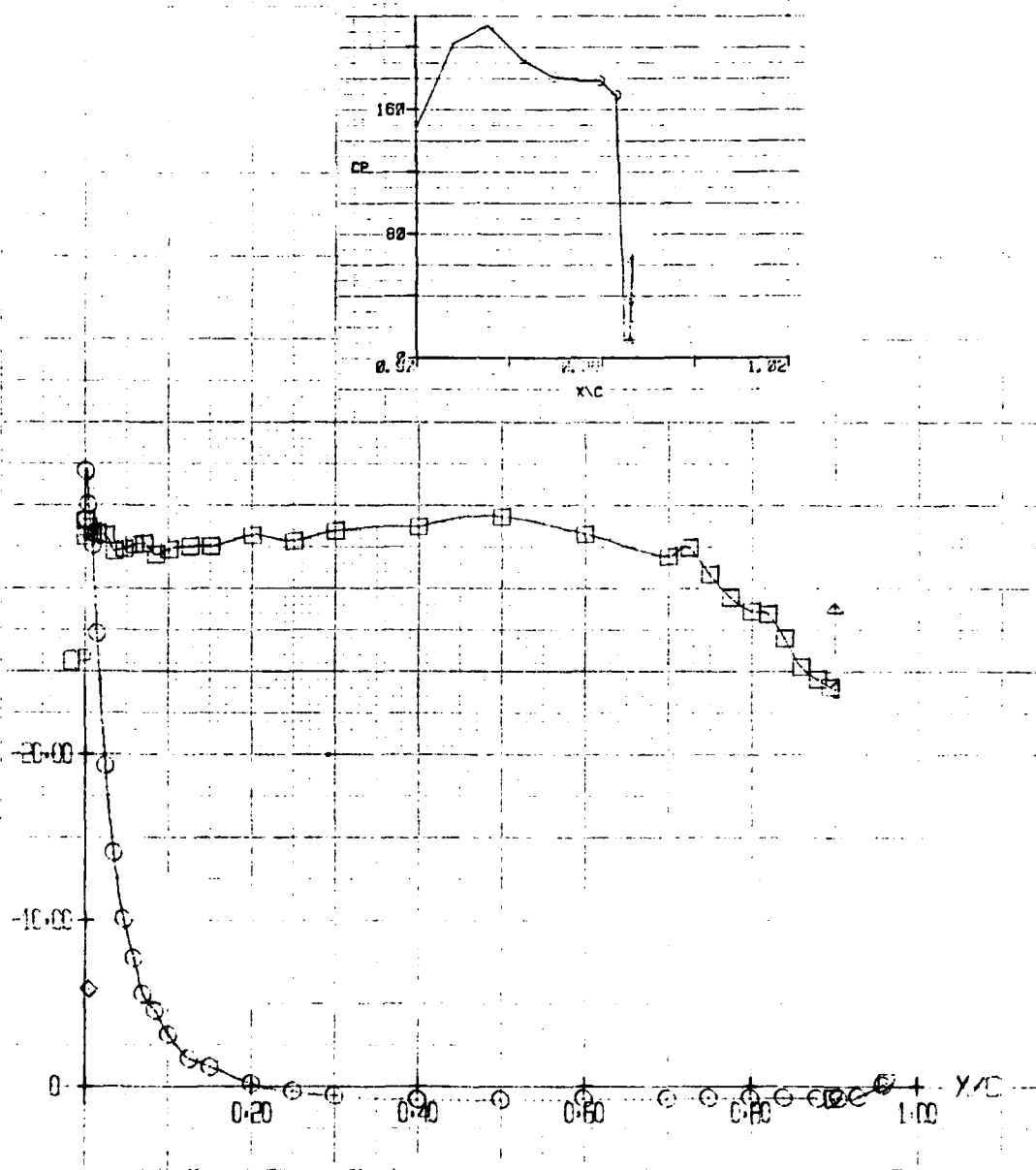


Figure I-42. $C_u = 15.22$, $\Theta = 90^\circ$, $\alpha = 0^\circ$ (Test 272, Run 51)
 $Q = 1.52$ psf

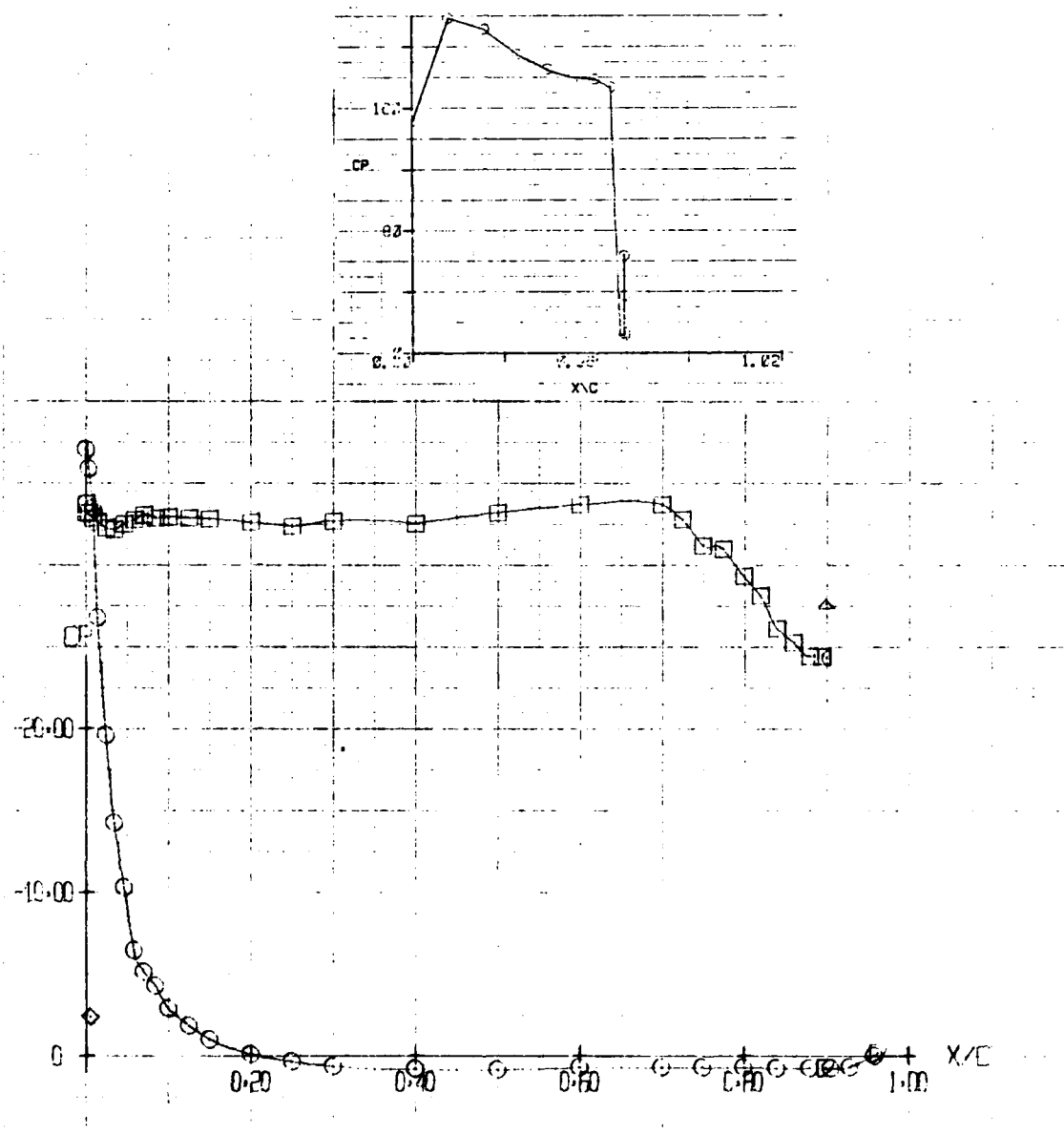


Figure I-43. $C_u = 15.34$, $\Theta = 90^\circ$, $\alpha = 0^\circ$ (Test 272, Run 55)
 $Q = 1.51$ psf

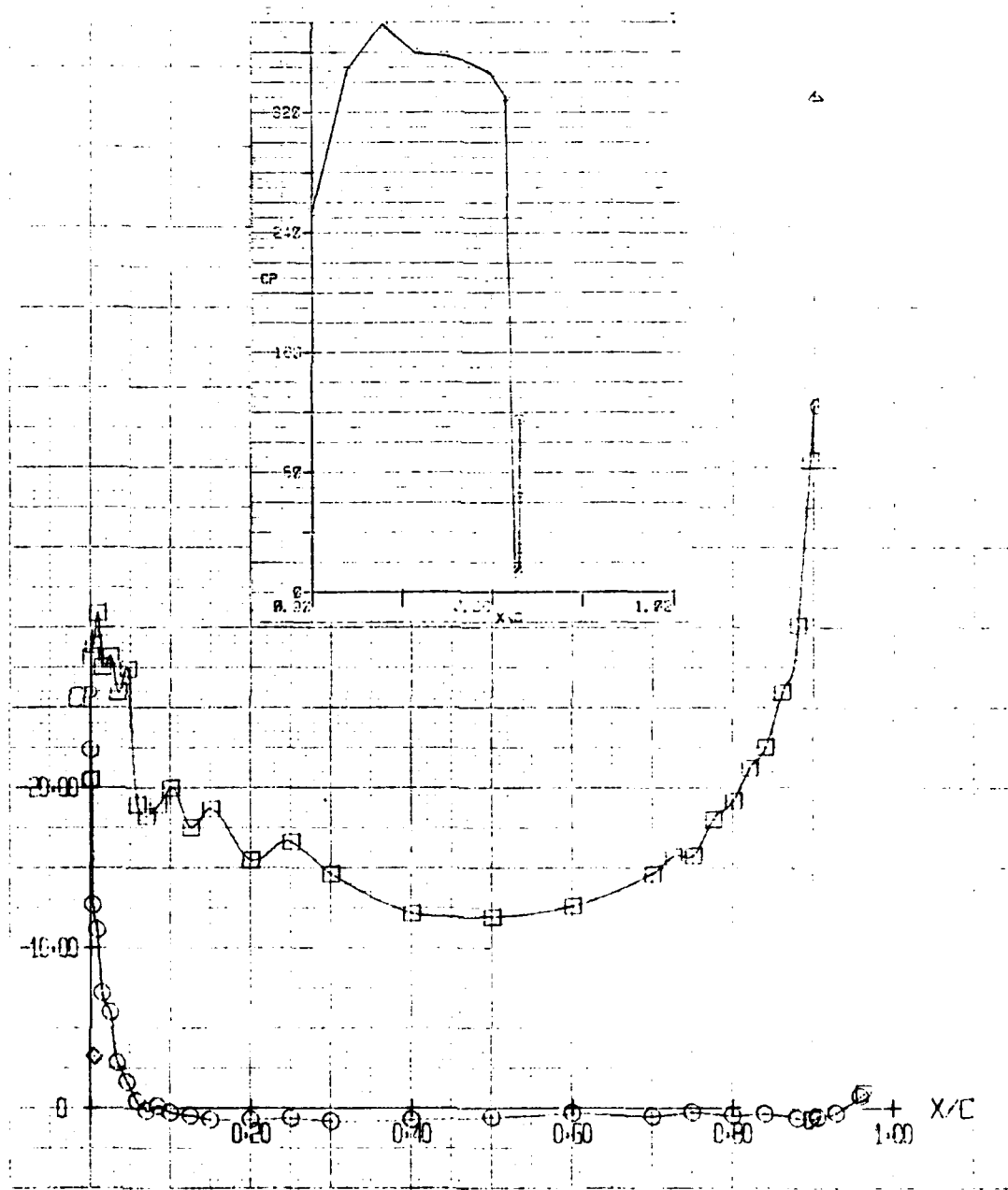


Figure I-44. $C_{\mu} = 25.53$, $\Theta = 90^\circ$, $\alpha = 0^\circ$ (Test 272, Run 52)
 $Q = 0.90$ psf

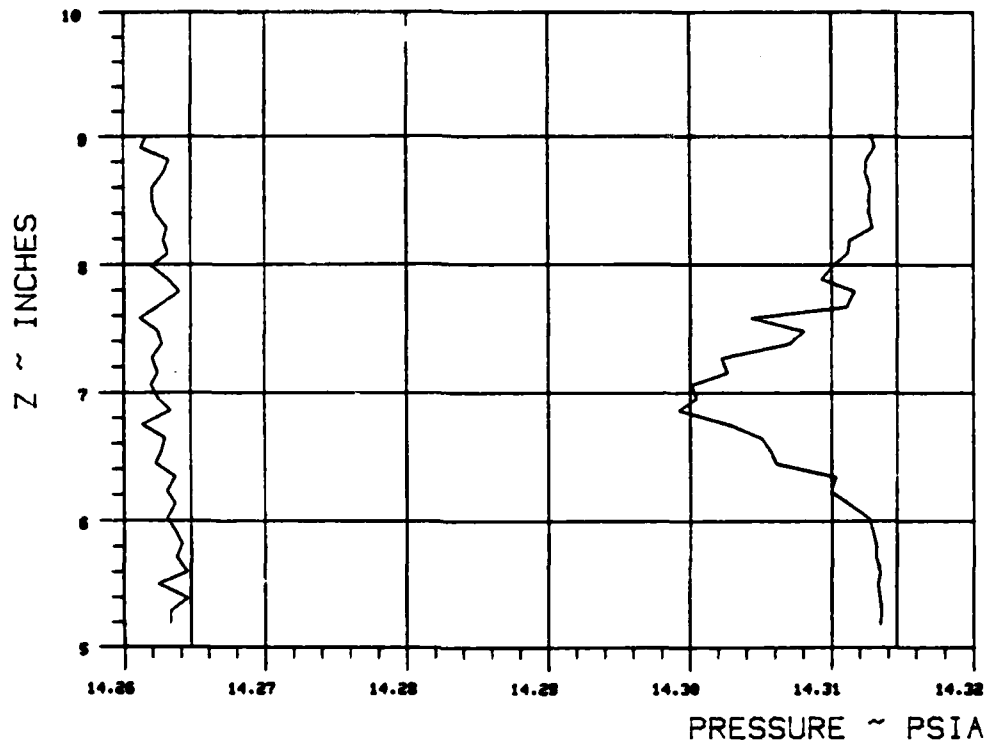
APPENDIX II

WAKE SURVEY DATA

WAKE SURVEY DATA

TEST 272 RUN 2

ANGLE OF ATTACK -10 NOMINAL JET ANGLE — C MU —



$X_{ee}/\text{chord} = \text{—}$ $Y_{ee}/\text{chord} = \text{—}$

BETA = —

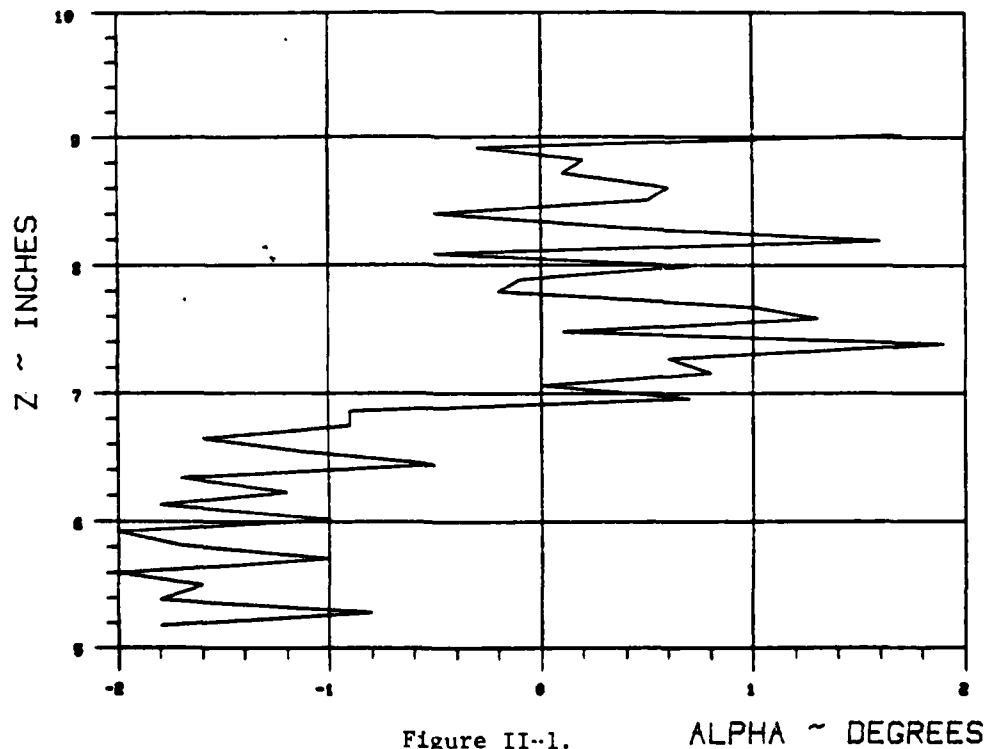
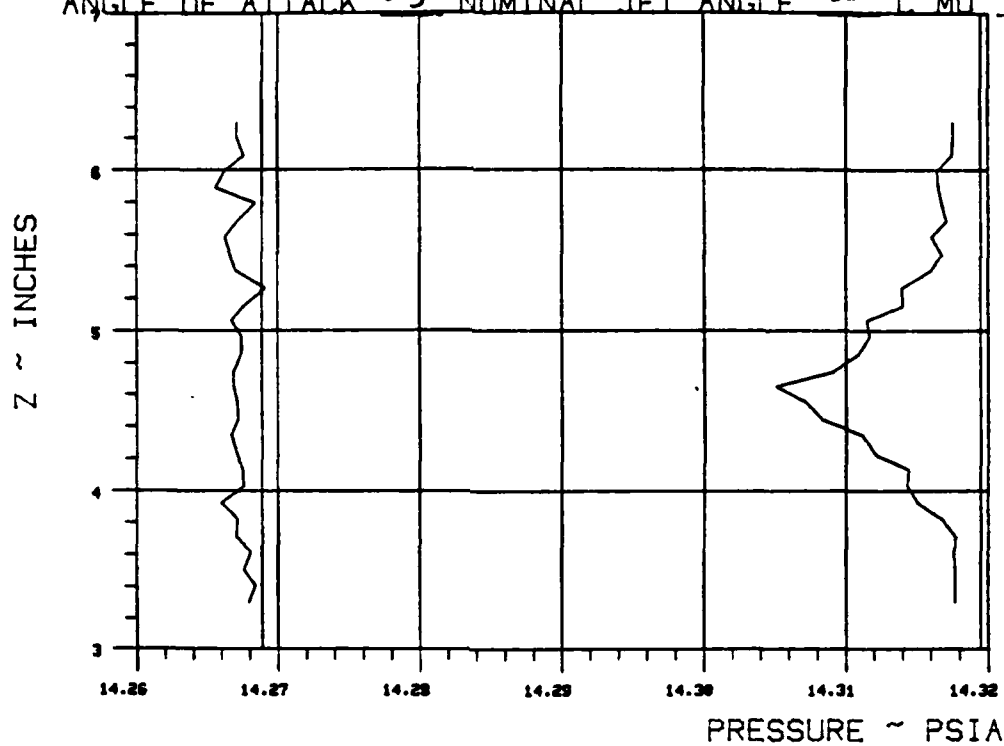


Figure II-1.

WAKE SURVEY DATA

TEST 272 RUN 3

ANGLE OF ATTACK -5 NOMINAL JET ANGLE - C MU



$X_{ss}/\text{chord} = \underline{\hspace{1cm}}$ $Y_{ss}/\text{chord} = \underline{\hspace{1cm}}$

BETA =

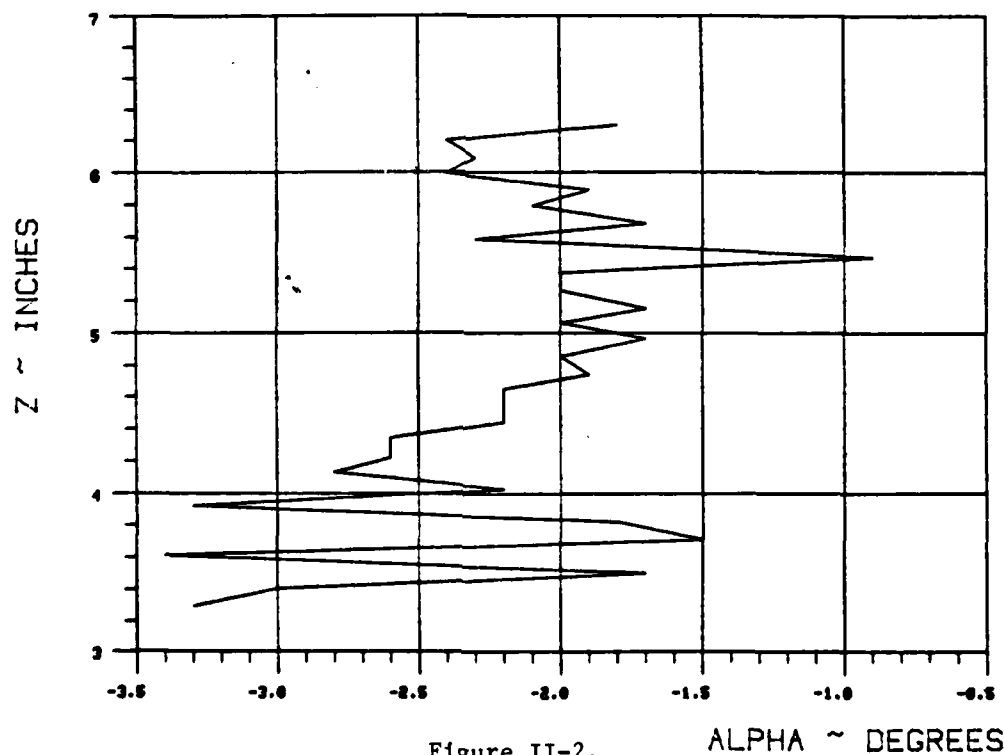
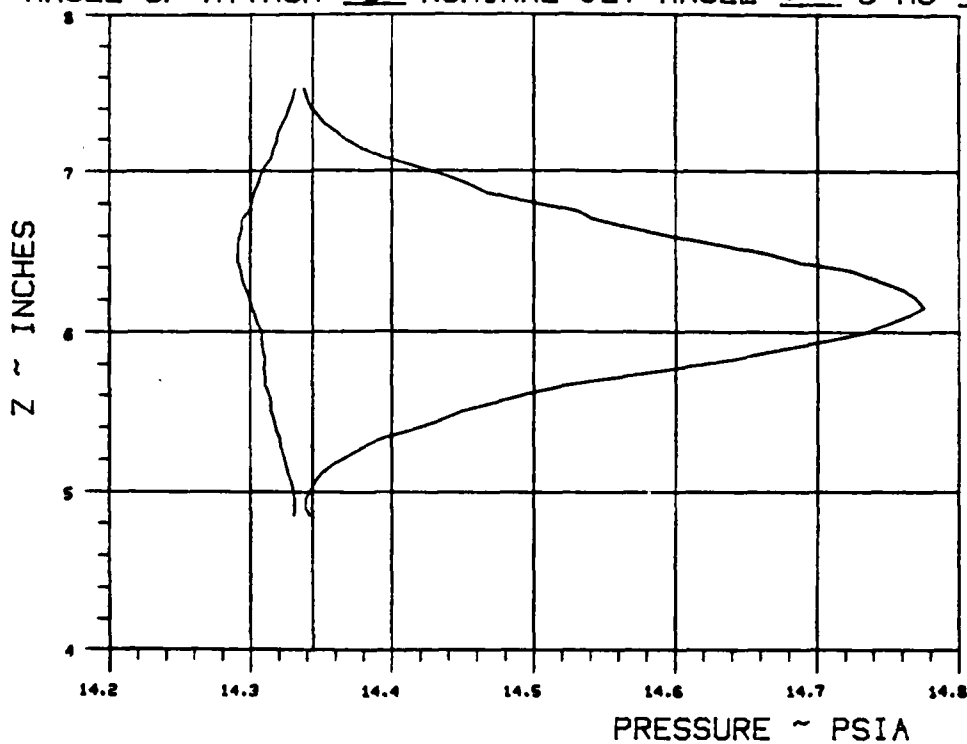


Figure II-2.

WAKE SURVEY DATA

TEST 272 RUN 12

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU —



$$X_{ss}/\text{chord} = .350 / Y_{ss}/\text{chord} = .1014$$

$$\text{BETA} = 20.83$$

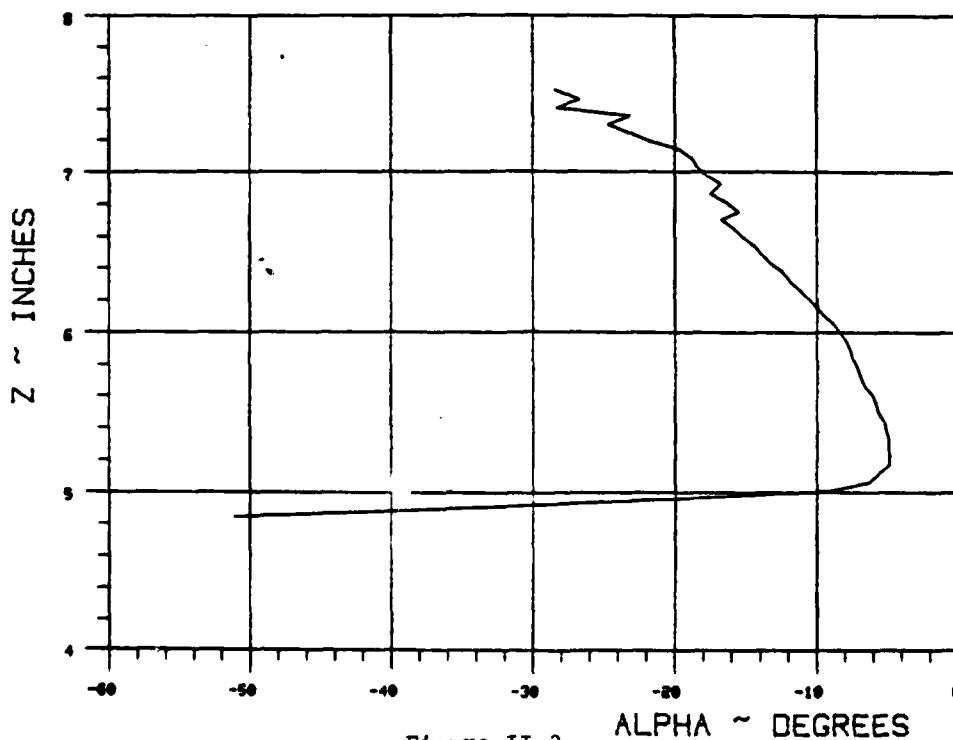
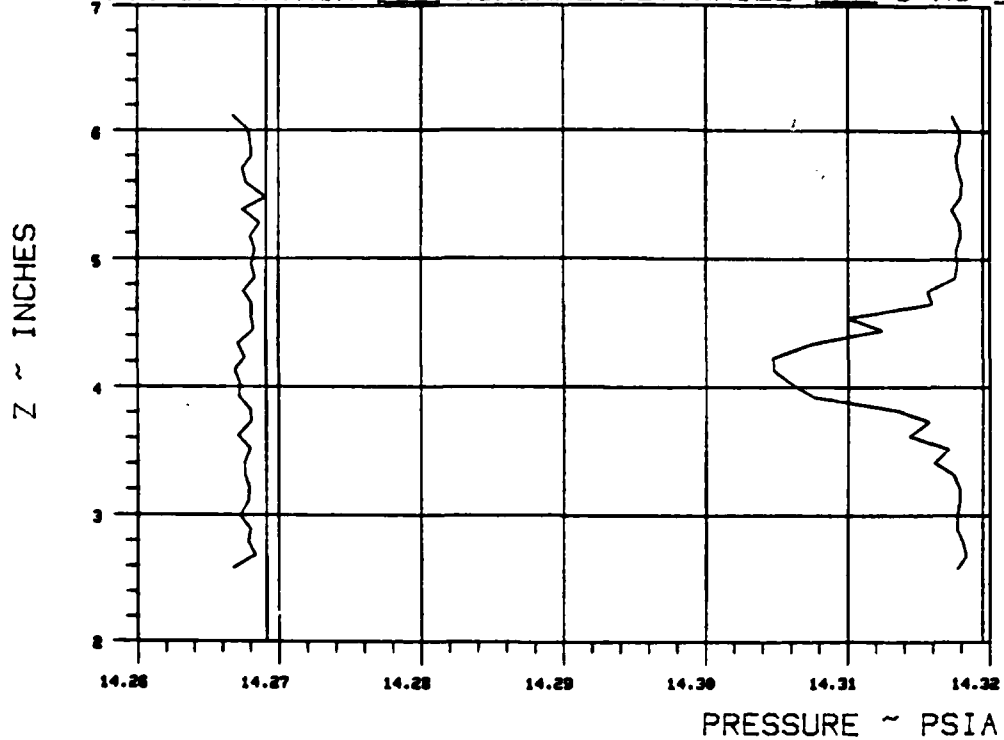


Figure II-3.

WAKE SURVEY DATA

TEST 272 RUN 4

ANGLE OF ATTACK 5 NOMINAL JET ANGLE — C MU —



X_{ss}/chord = — Y_{ss}/chord = —

BETA = —

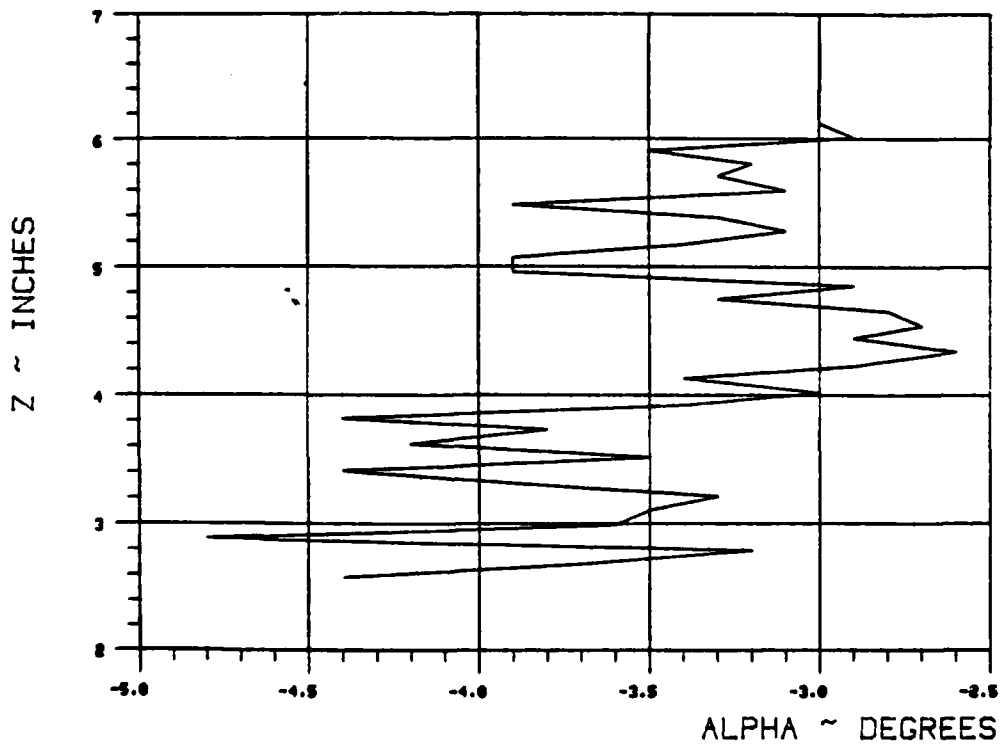
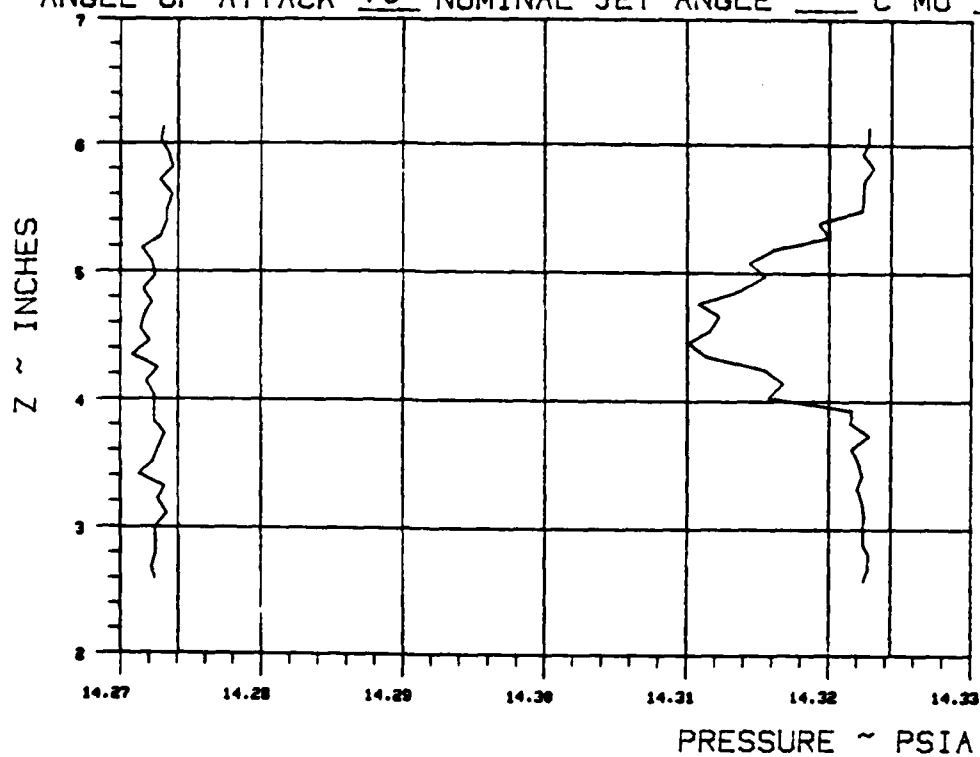


Figure II-4.

WAKE SURVEY DATA

TEST 212 RUN 5

ANGLE OF ATTACK 10 NOMINAL JET ANGLE — C MU —



X_{ss}/chord = — Y_{ss}/chord = —

BETA = —

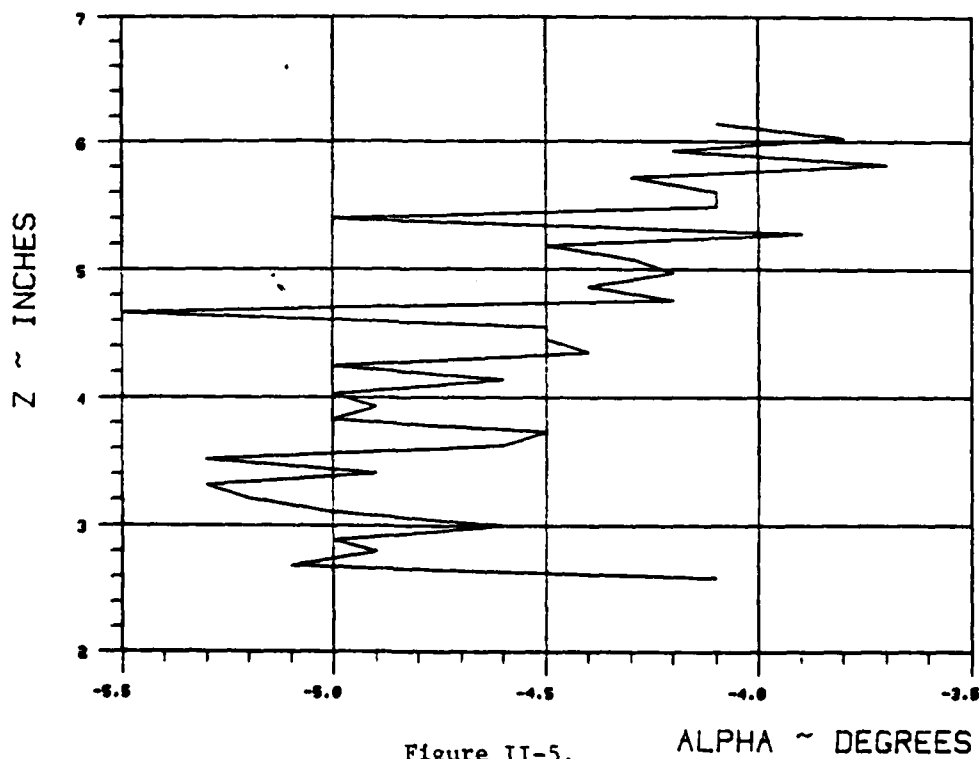
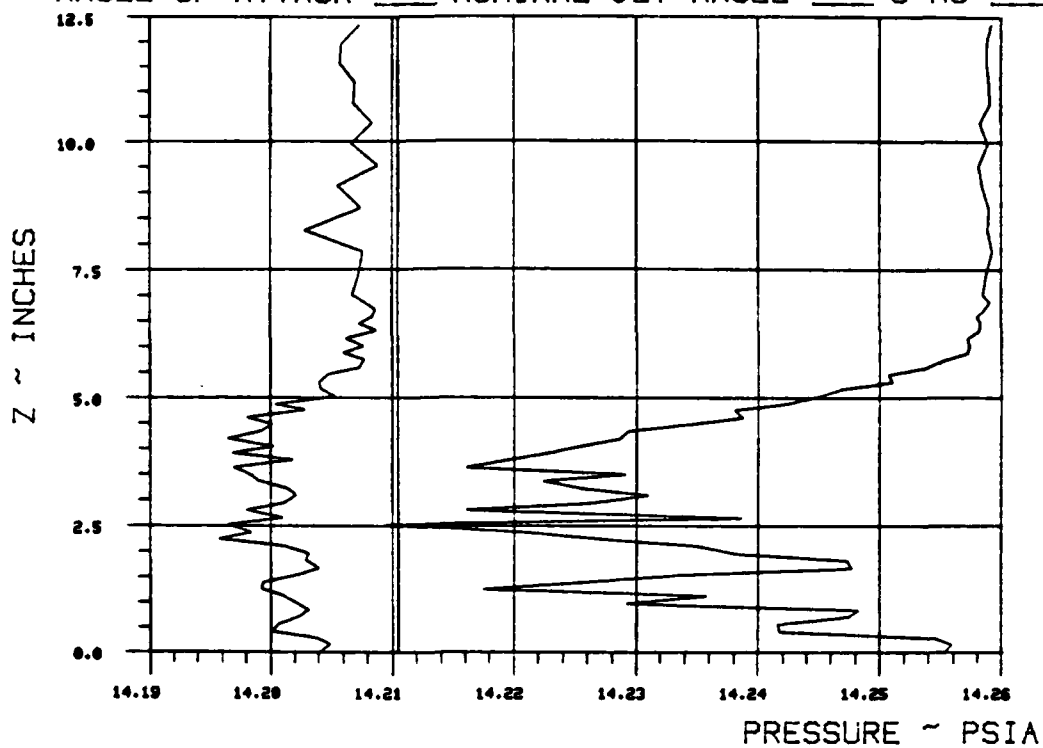


Figure II-5.

WAKE SURVEY DATA

TEST 272 RUN 8

ANGLE OF ATTACK 15 NOMINAL JET ANGLE — C MU —



X_{ss}/chord = — Y_{ss}/chord = —

BETA = —

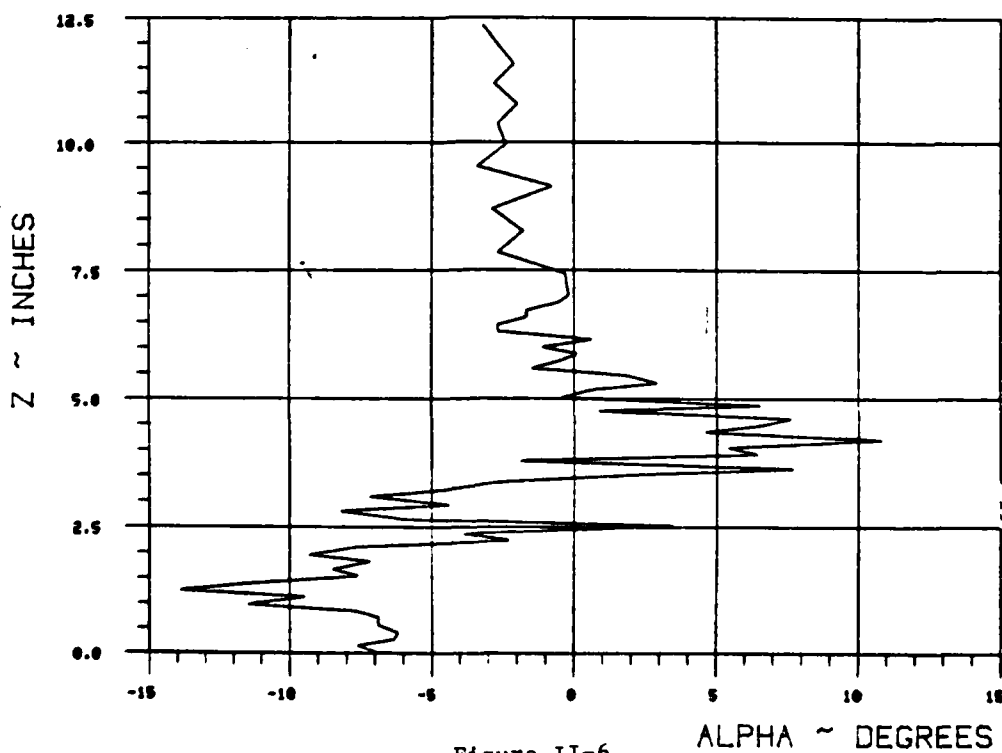
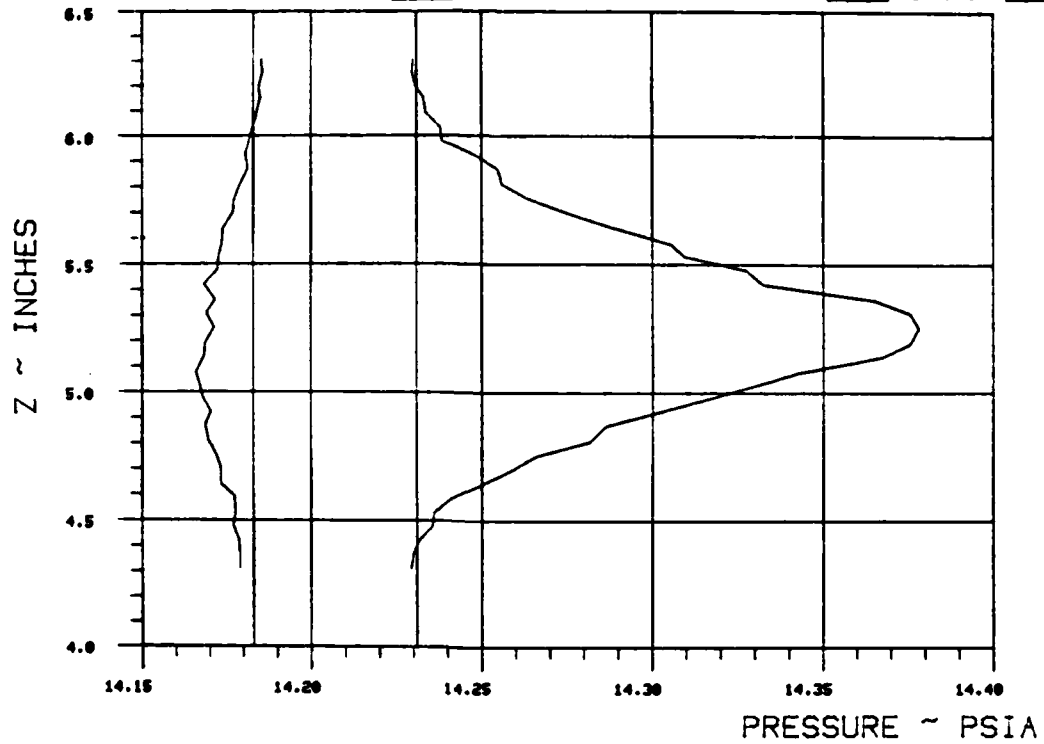


Figure II-6.

WAKE SURVEY DATA

TEST 272 RUN 9

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU .5327



$$X_{ee}/\text{chord} = .3733 \quad Y_{ee}/\text{chord} = .1415$$

$$\text{BETA} = 20.83$$

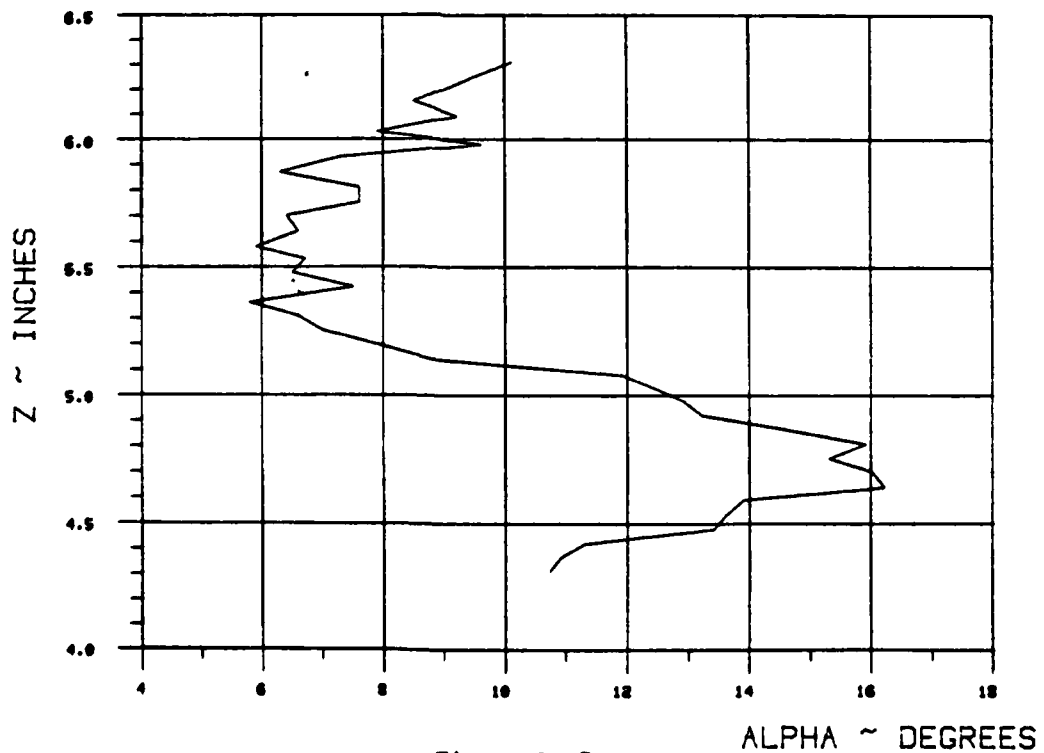
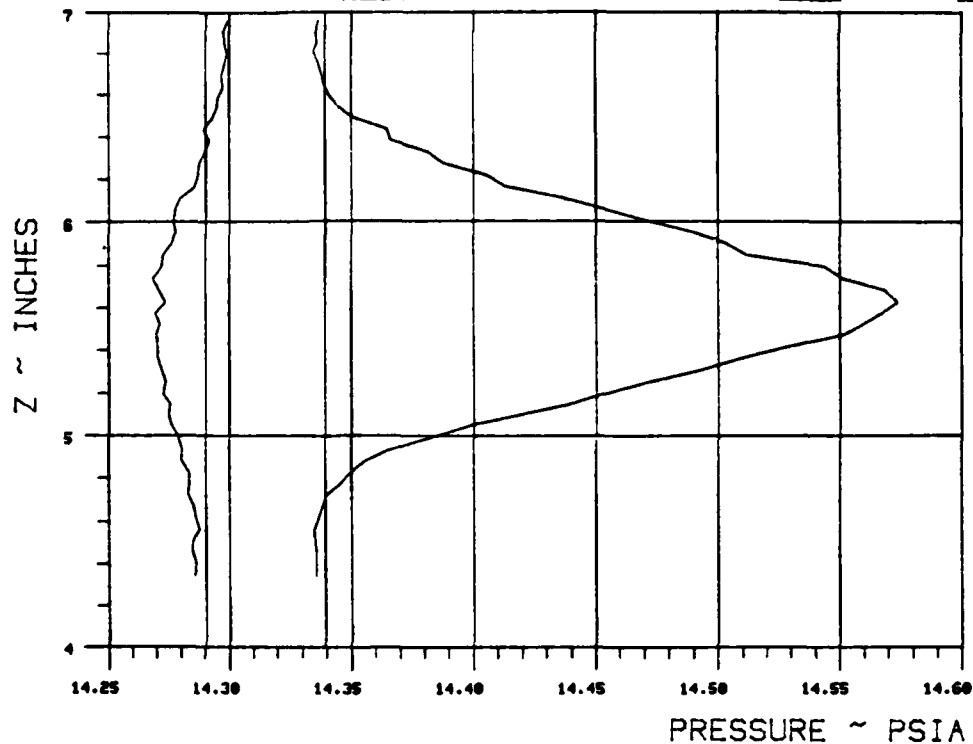


Figure II-7.

WAKE SURVEY DATA

TEST 272 RUN 10

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 1.07



$X_{ss}/\text{chord} = .374$ $Y_{ss}/\text{chord} = .1433$

BETA = 20.83

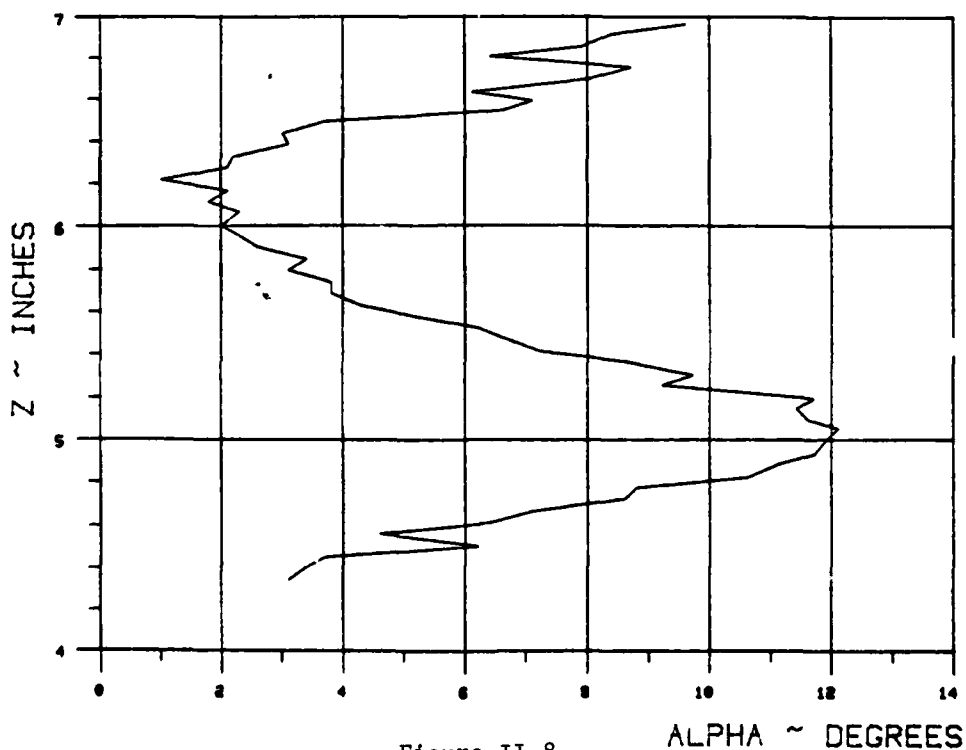
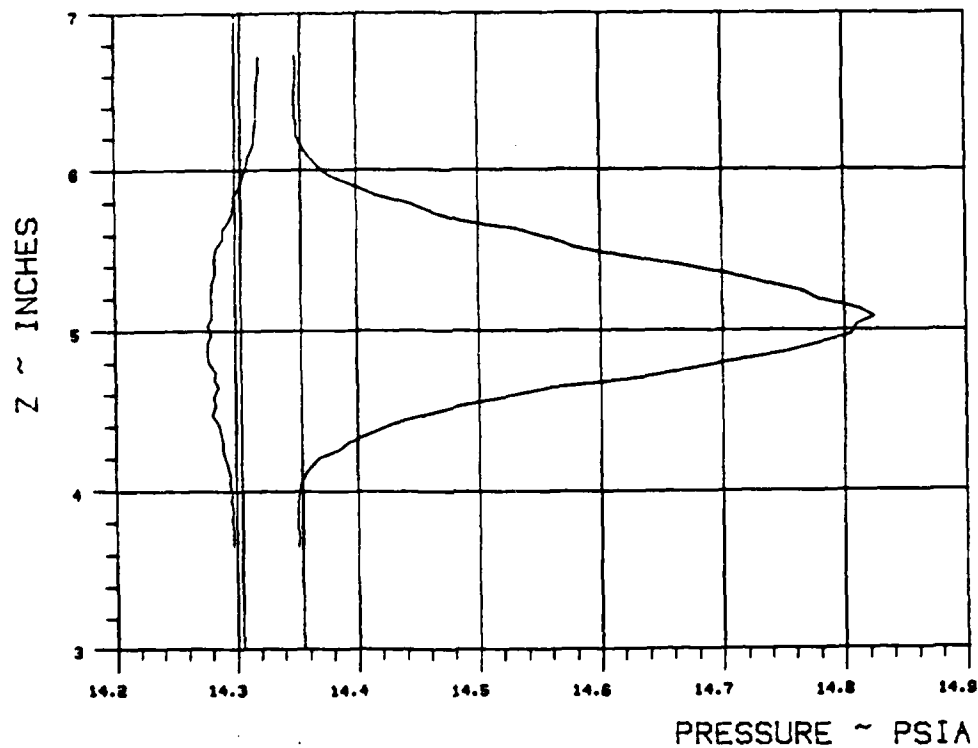


Figure II-8.

WAKE SURVEY DATA

TEST 272 RUN 11

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 2.1154



$X_{ss}/\text{chord} = .3581$ $Y_{ss}/\text{chord} = .1015$

BETA = 20.83

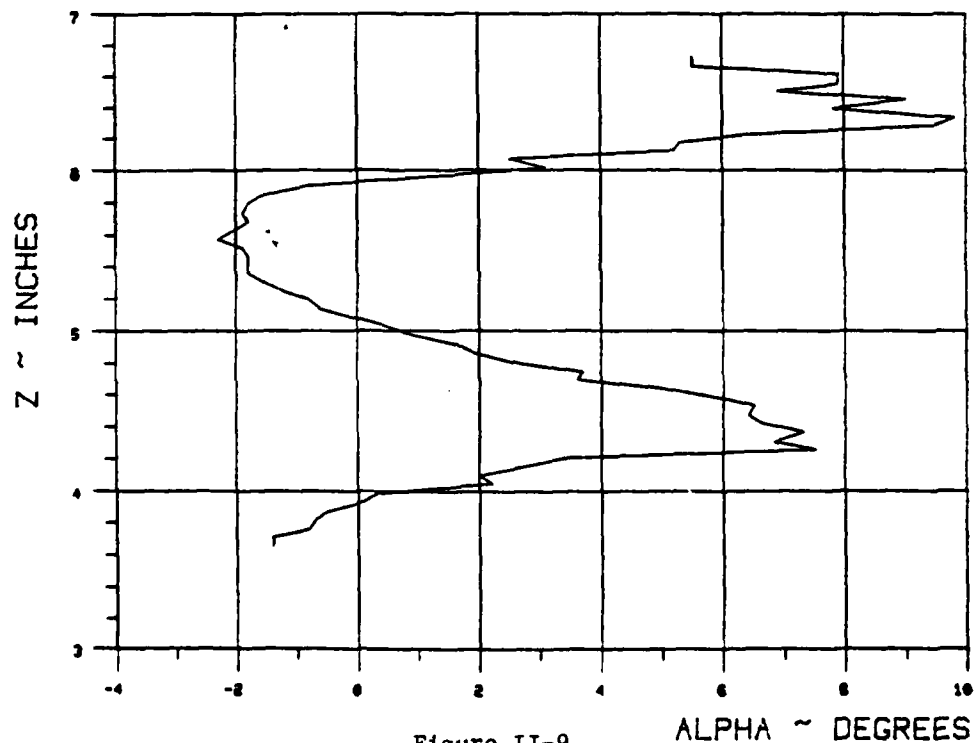
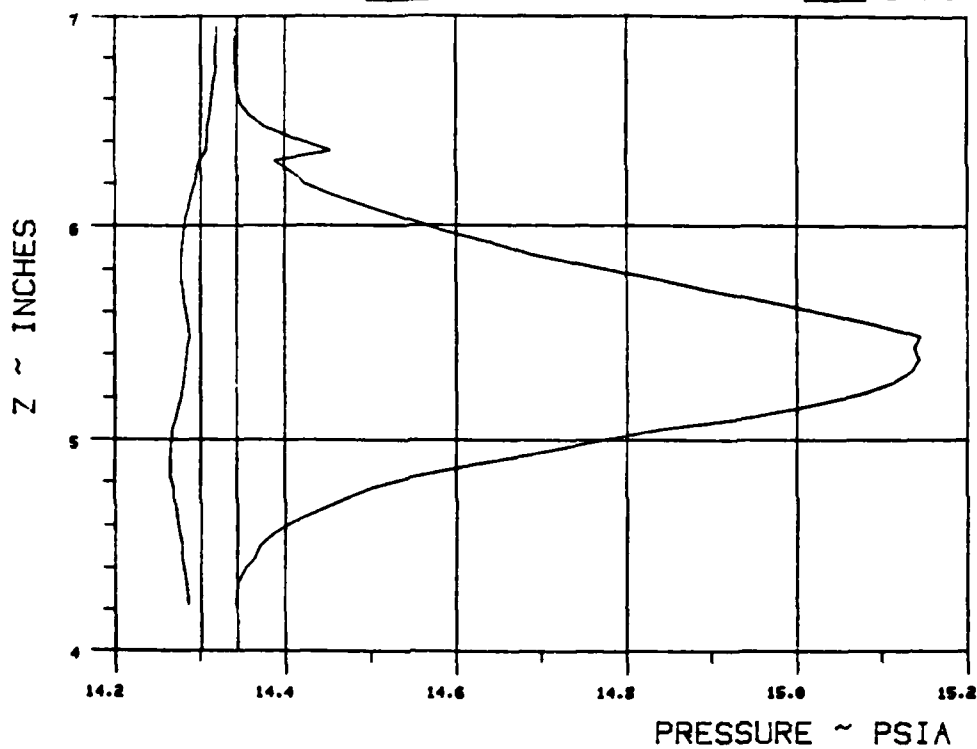


Figure II-9.

WAKE SURVEY DATA

TEST 272 RUN 13

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 4.2601



$X_{ee}/chord = .3581$ $Y_{ee}/chord = .1016$

BETA = 20.83

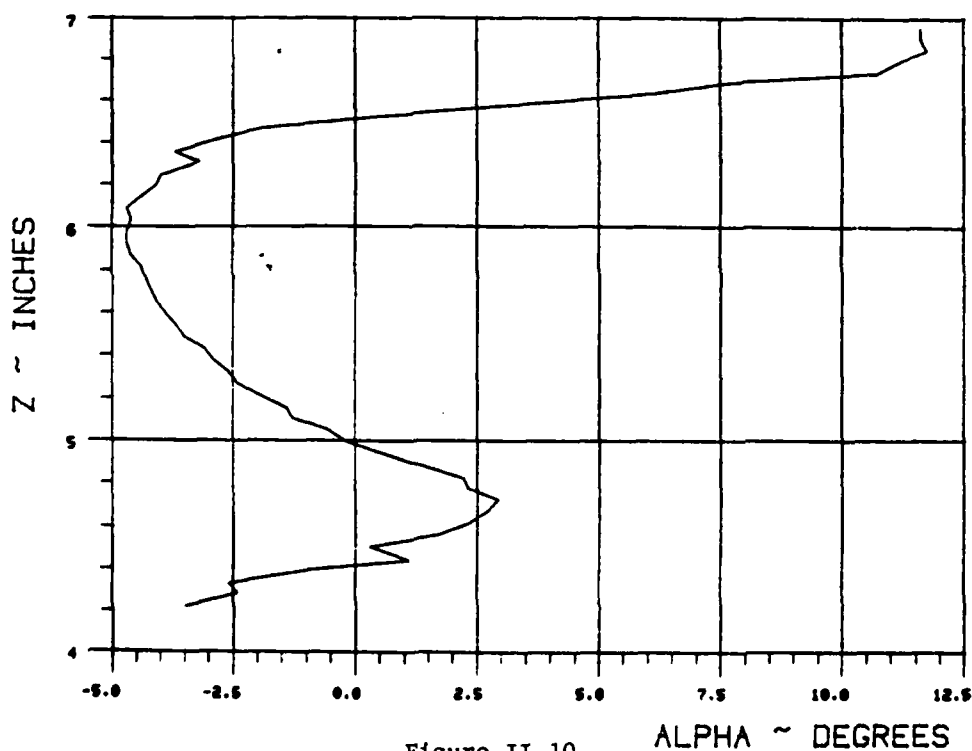
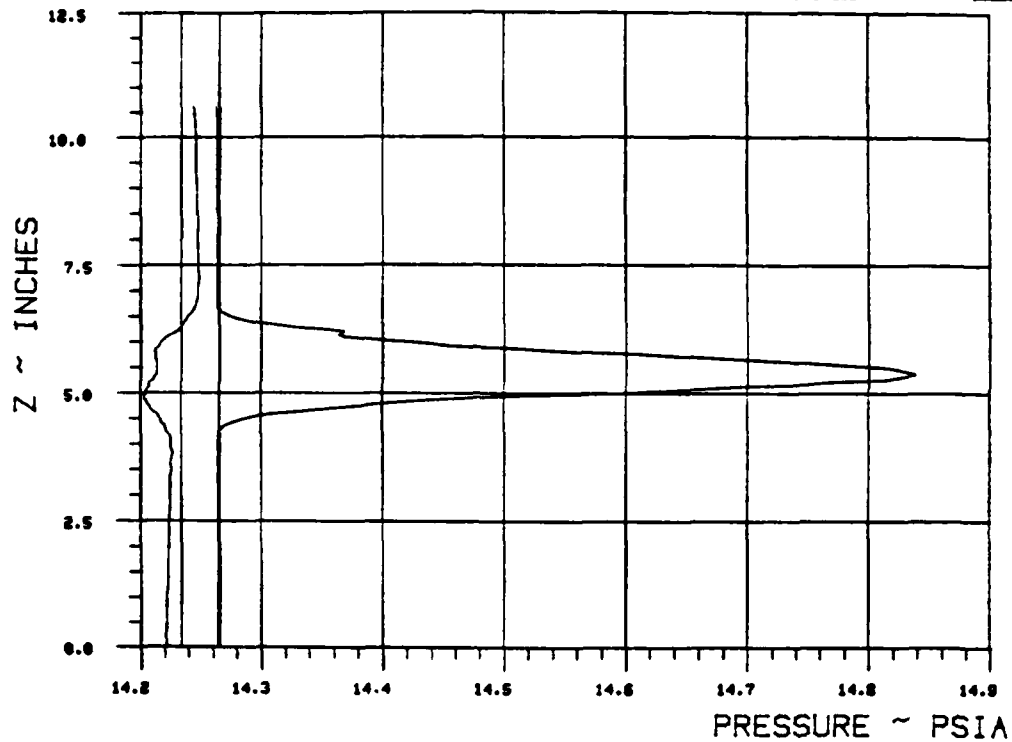


Figure II-10.

WAKE SURVEY DATA

TEST 272 RUN 14

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 4.2694



$X_{ss}/\text{chord} = .3587$ $Y_{ss}/\text{chord} = .1032$

BETA = 20.83

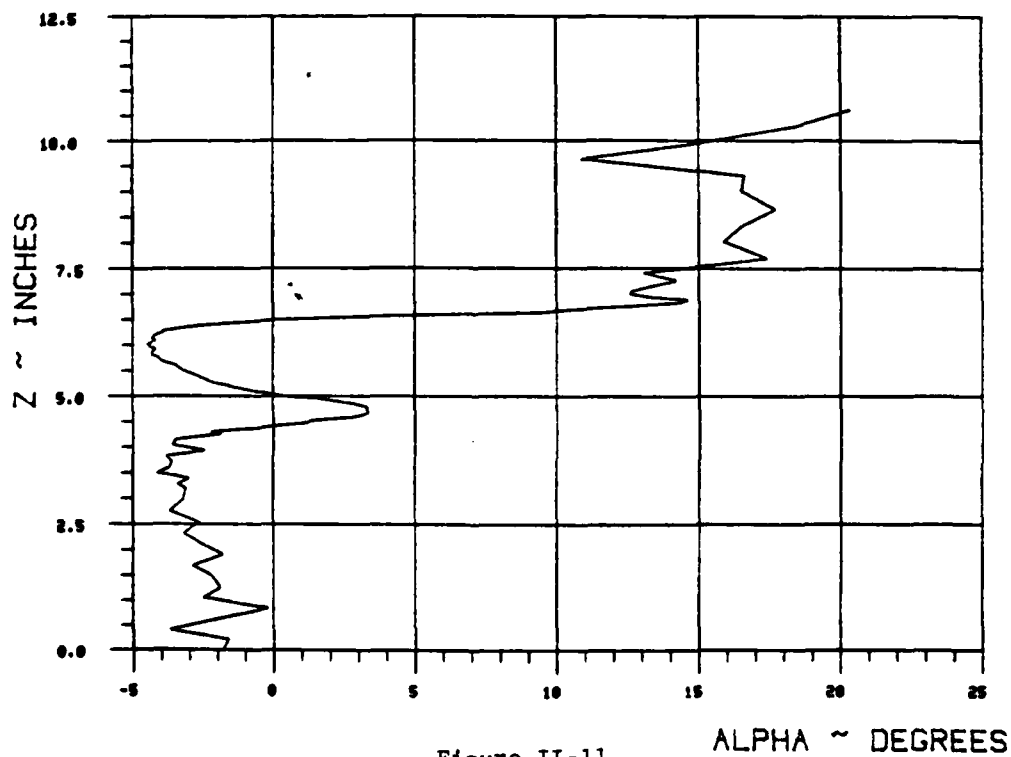
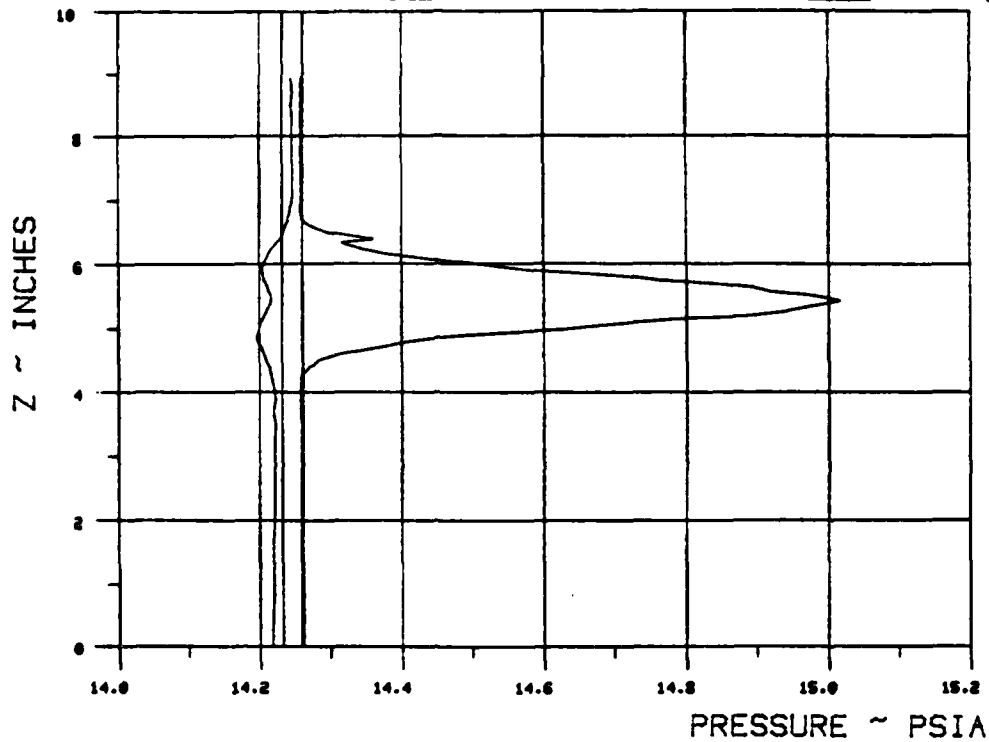


Figure II-11.

WAKE SURVEY DATA

TEST 272 RUN 15

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 6.4062



$X_{ss}/\text{chord} = .3564$ $Y_{ss}/\text{chord} = .097$

BETA = 20.83

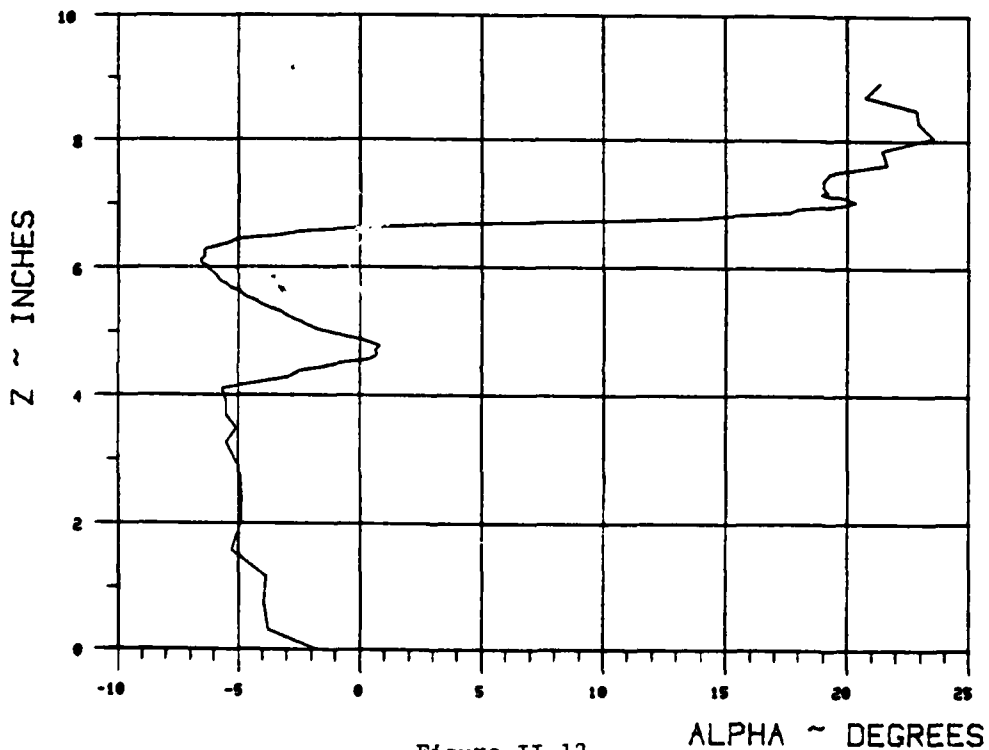
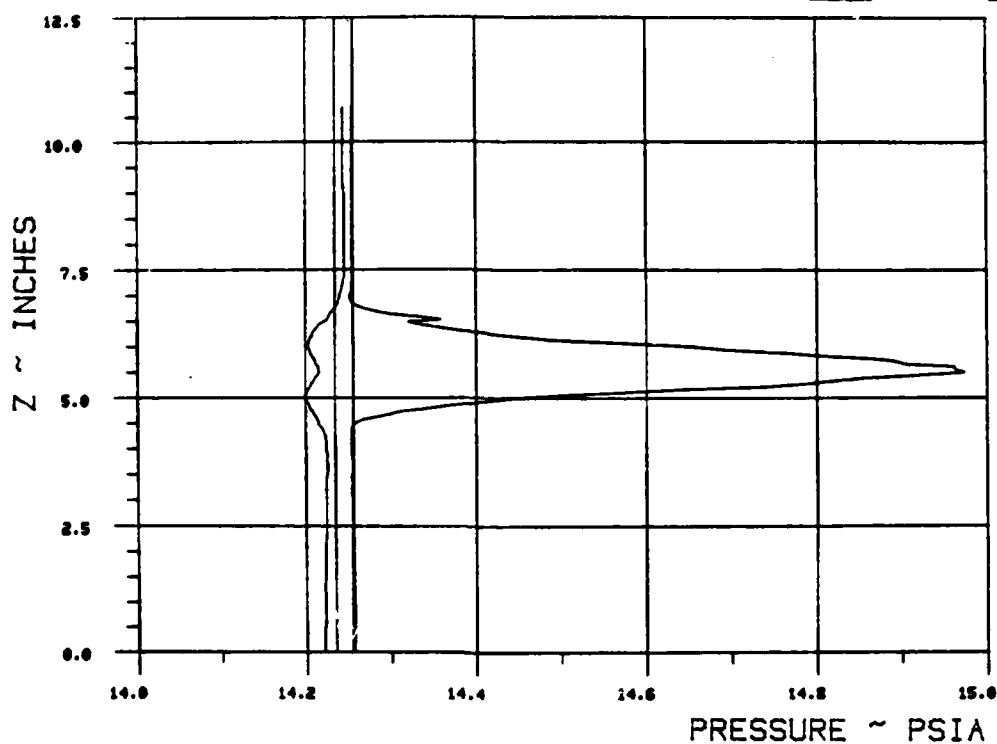


Figure II-12.

WAKE SURVEY DATA

TEST 272 RUN 16

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 8.4348



$X_{ee}/\text{chord} = .3569$ $Y_{ee}/\text{chord} = .0985$

BETA = 20.83

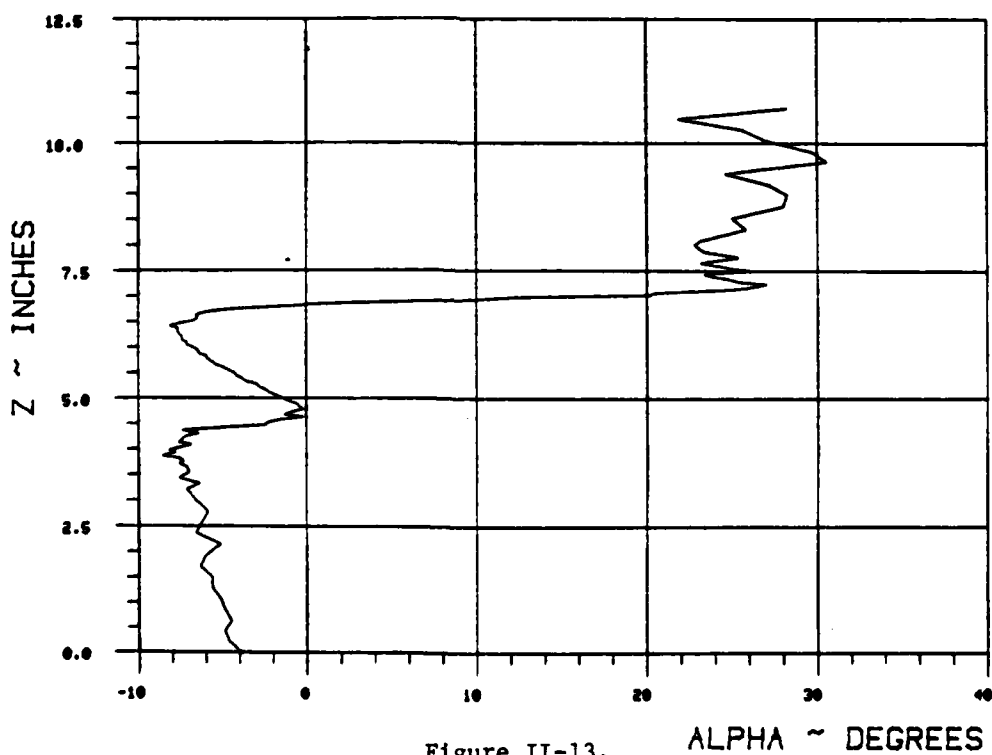
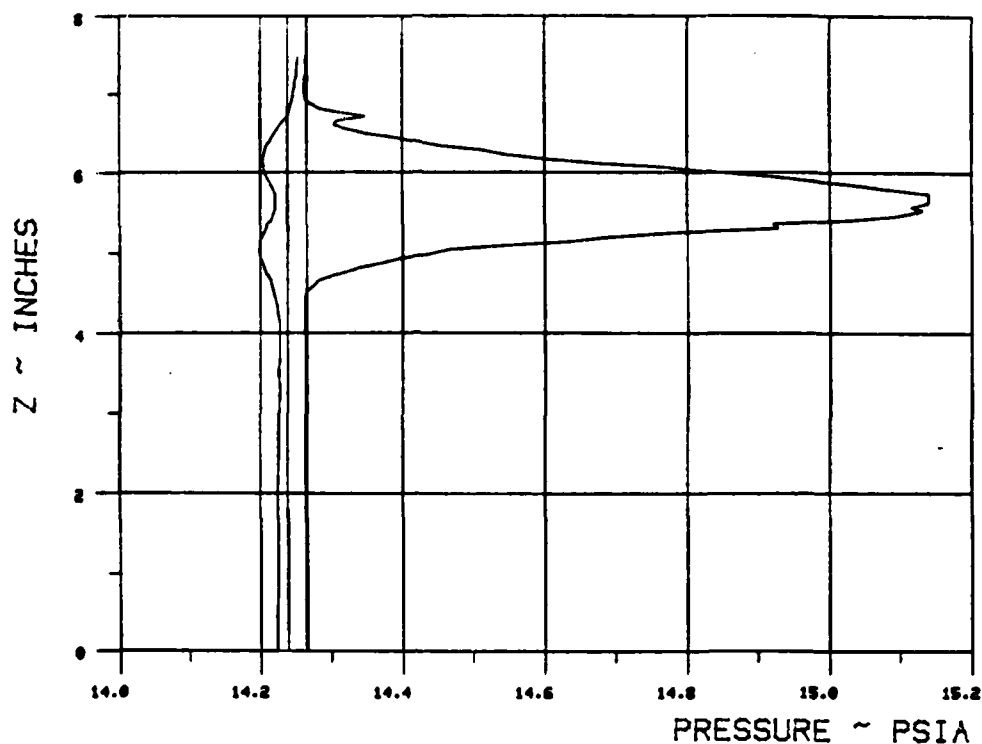


Figure II-13.

WAKE SURVEY DATA

TEST 272 RUN 17

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 8.6019



$X_{ee}/\text{cho. d} = .3578$ $Y_{ee}/\text{chord} = .1008$

BETA = 20.83

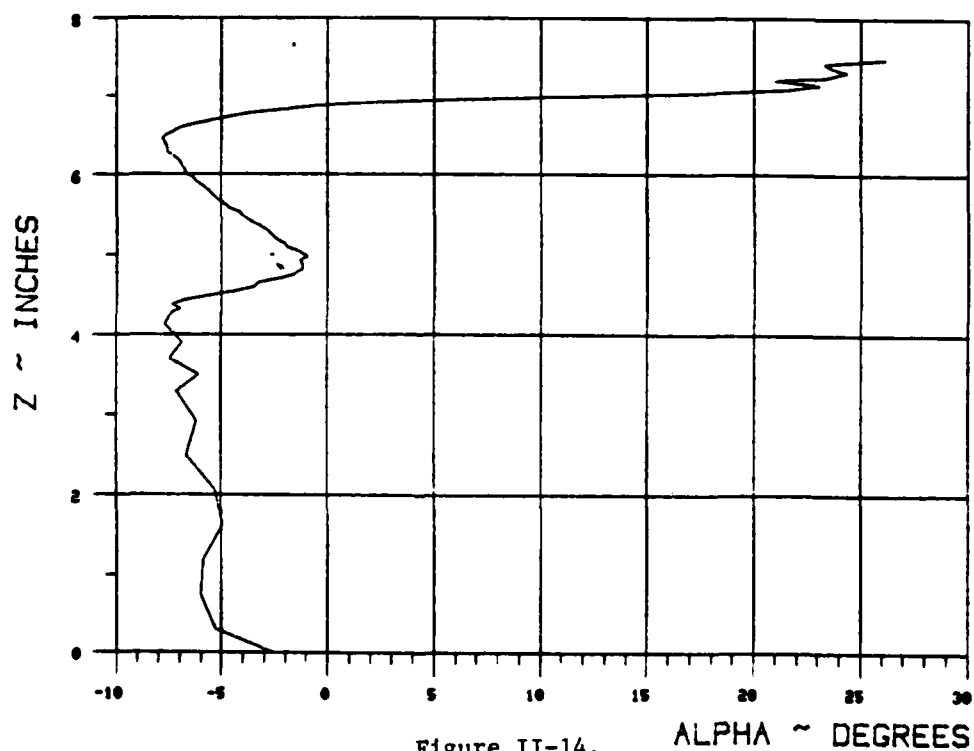
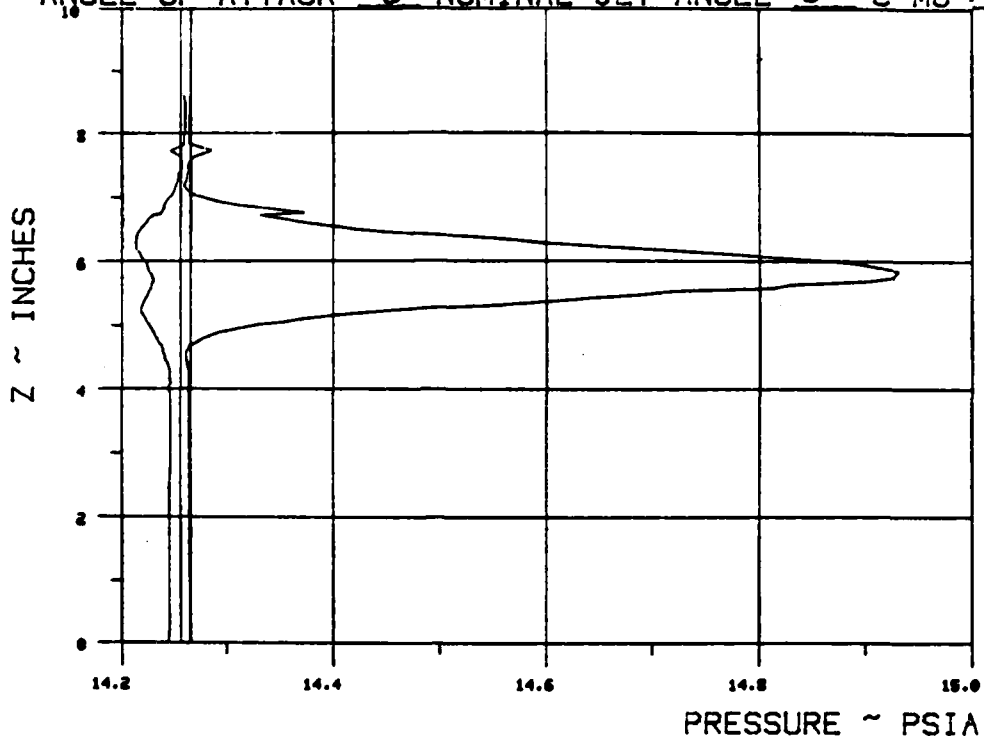


Figure II-14.

WAKE SURVEY DATA

TEST 272 RUN 1B

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 17.03/2



$X_{ss}/\text{chord} = .3578$ $Y_{ss}/\text{chord} = .1008$

BETA = 20.83

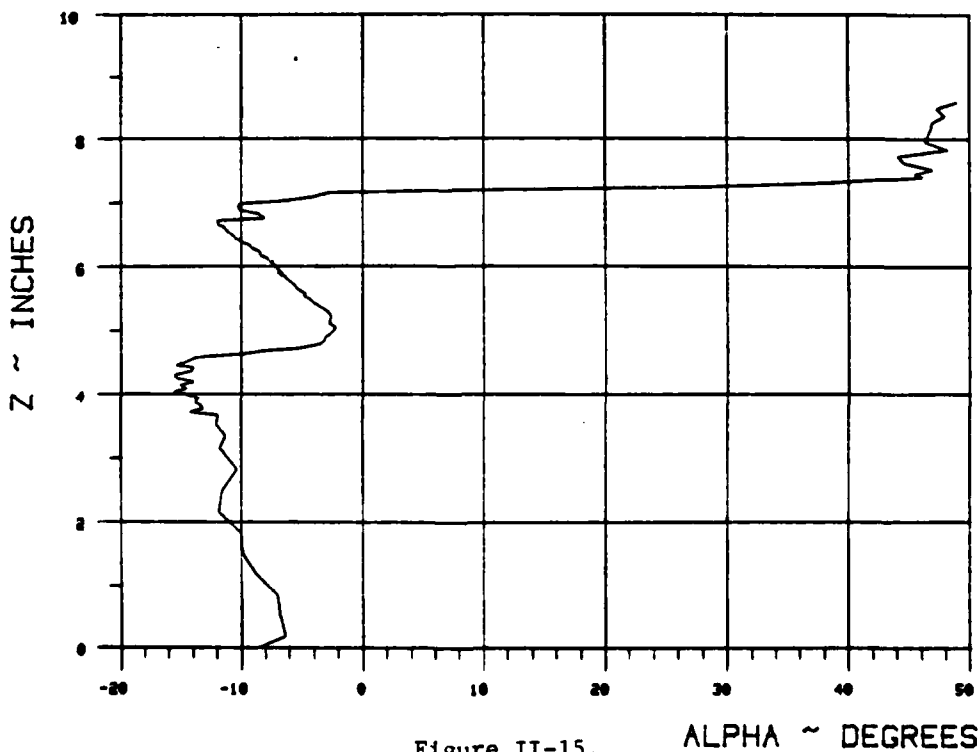
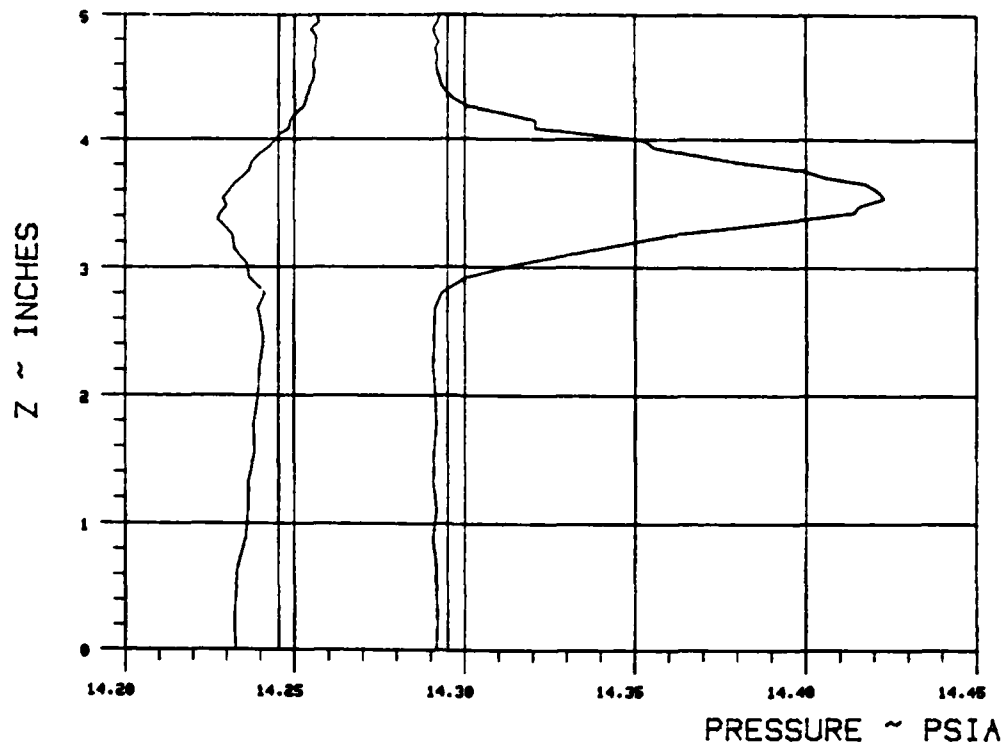


Figure II-15.

WAKE SURVEY DATA

TEST 272 RUN 19

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 50 C MU .5257



$X_{ee}/\text{chord} = .225$ $Y_{ee}/\text{chord} = .011$

BETA = 30

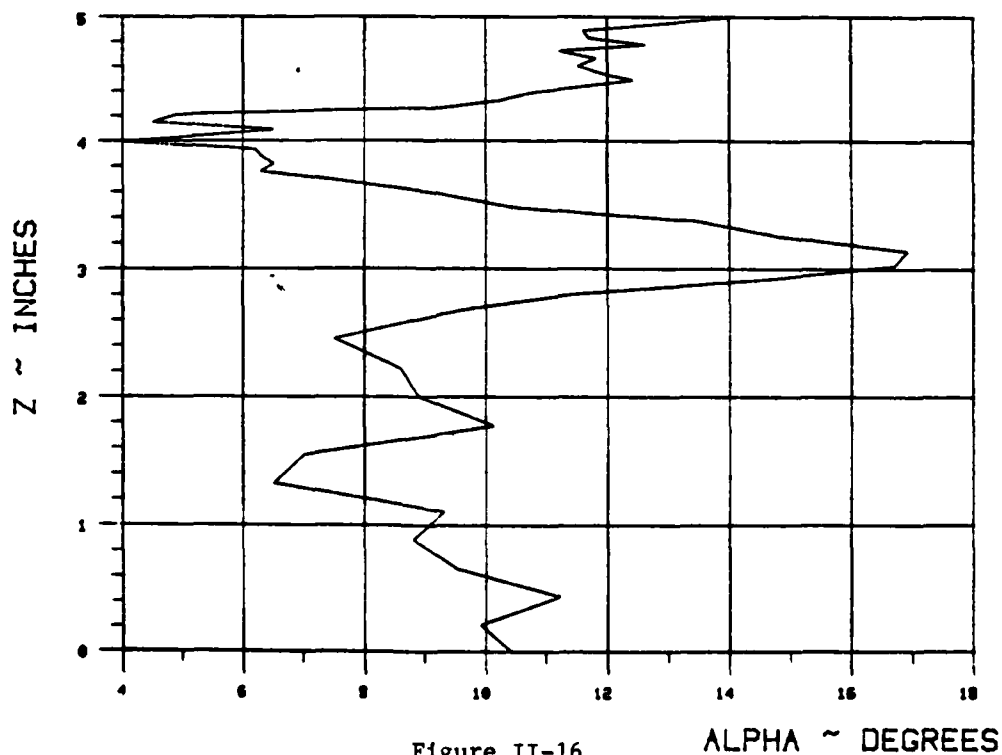
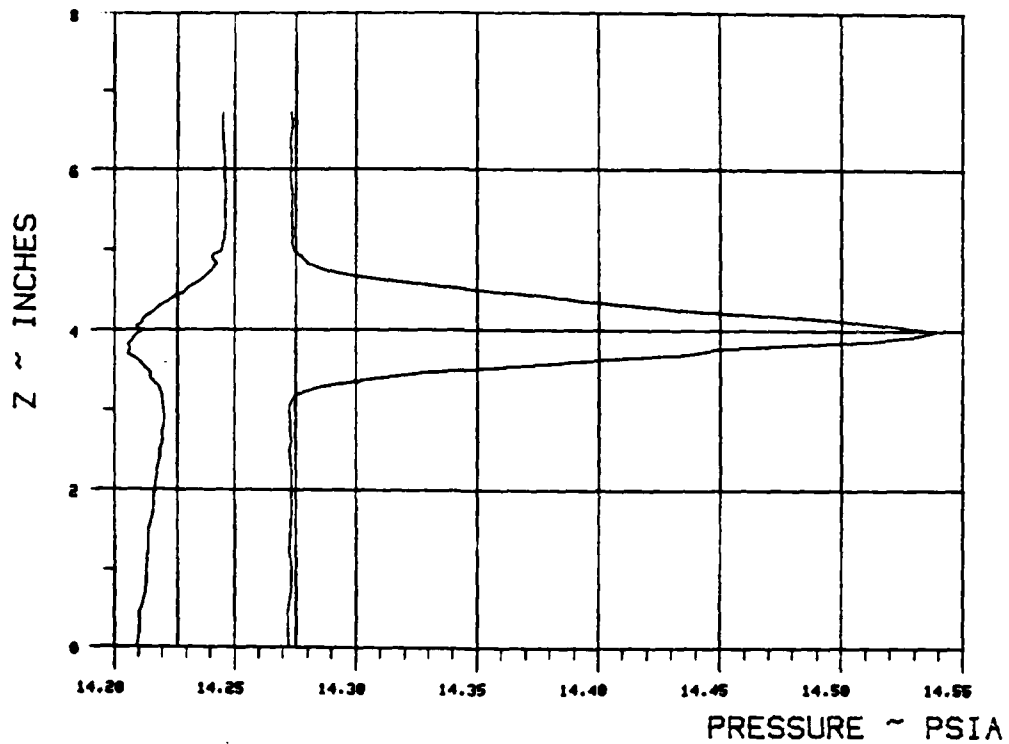


Figure II-16.

WAKE SURVEY DATA

TEST 272 RUN 20

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 50 C MU 1.0/3



$X_{ee}/\text{chord} = .2252$ $Y_{ee}/\text{chord} = .0113$

BETA = 30

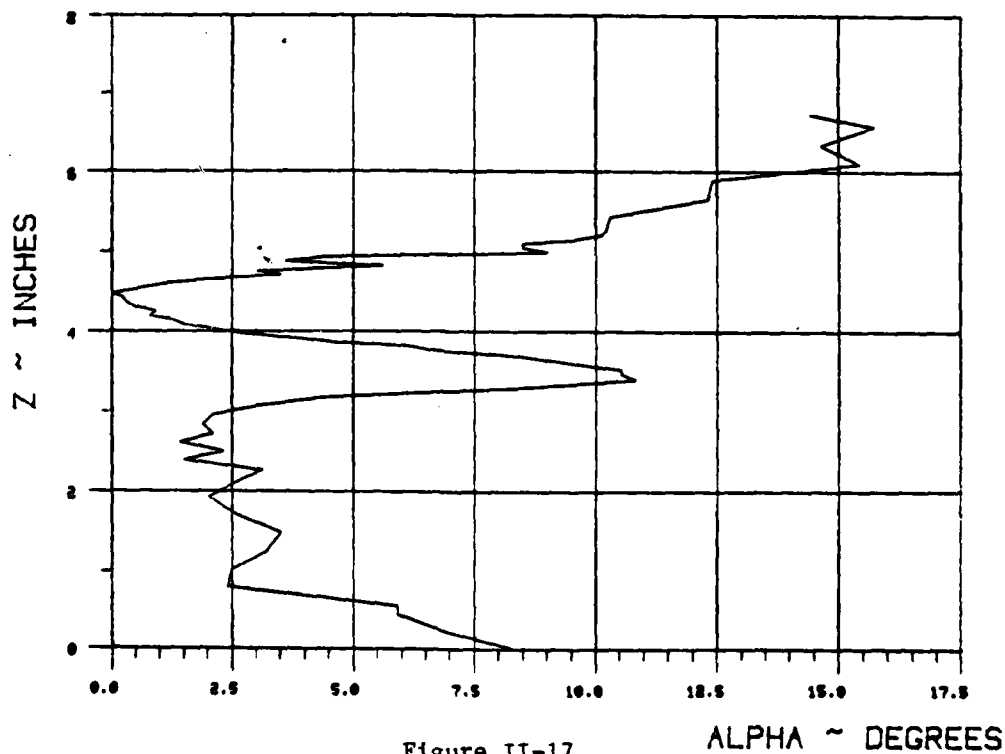
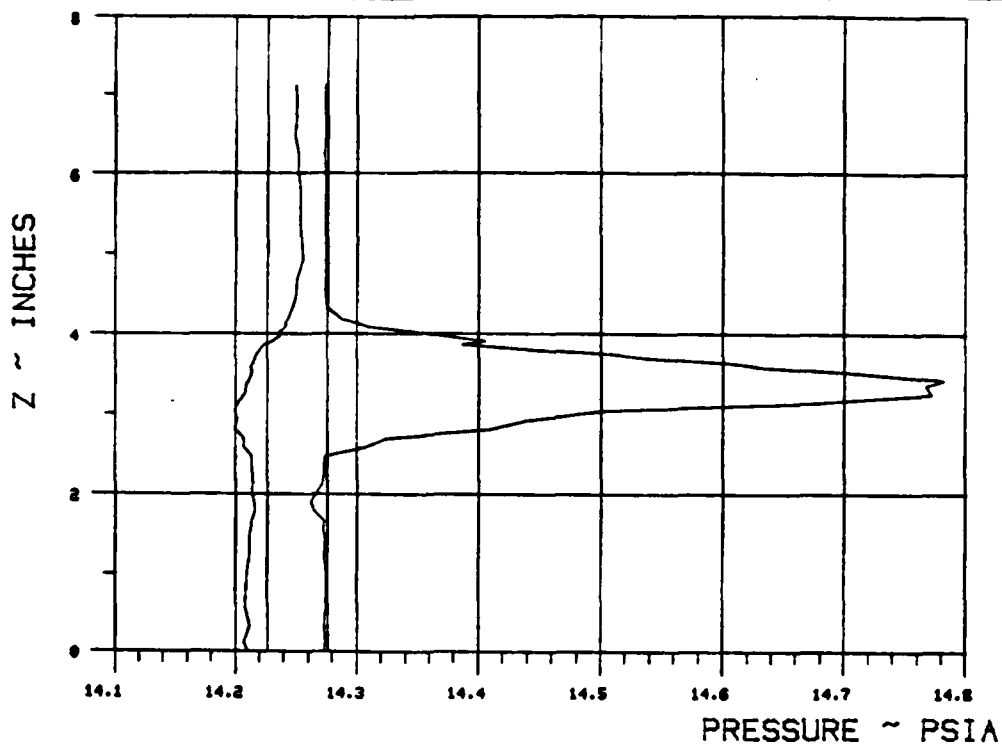


Figure II-17.

WAKE SURVEY DATA

TEST 272 RUN 21

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 50 C MU 2.045



$X_{ee}/\text{chord} = .2091$ $Y_{ee}/\text{chord} = -.0166$

BETA = 30

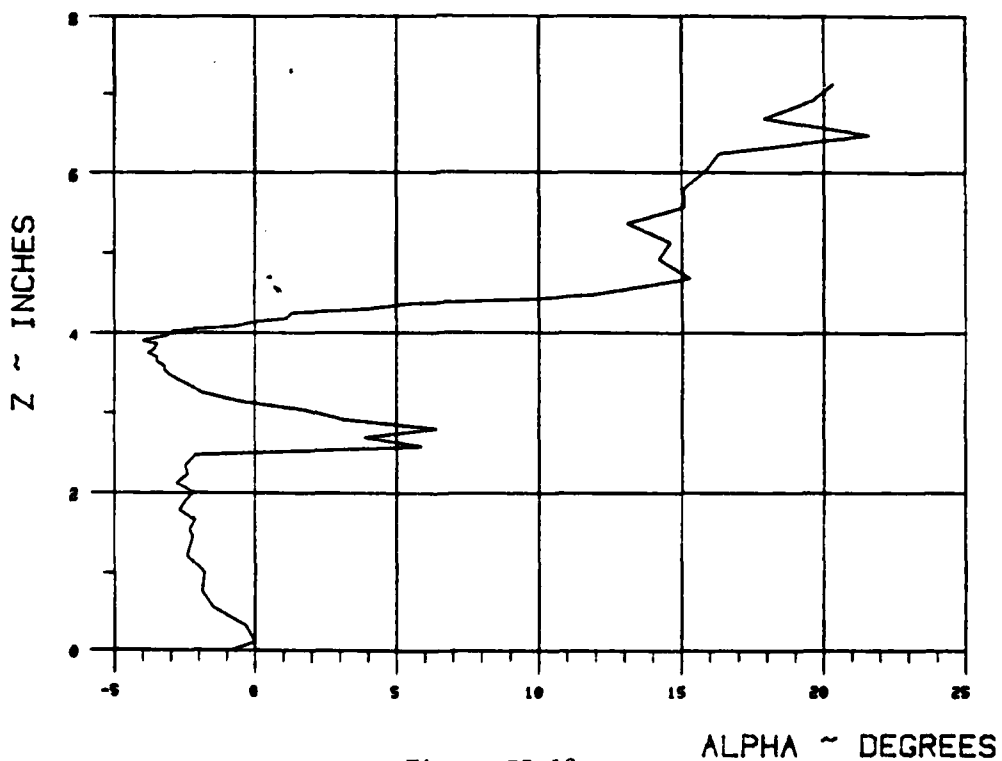
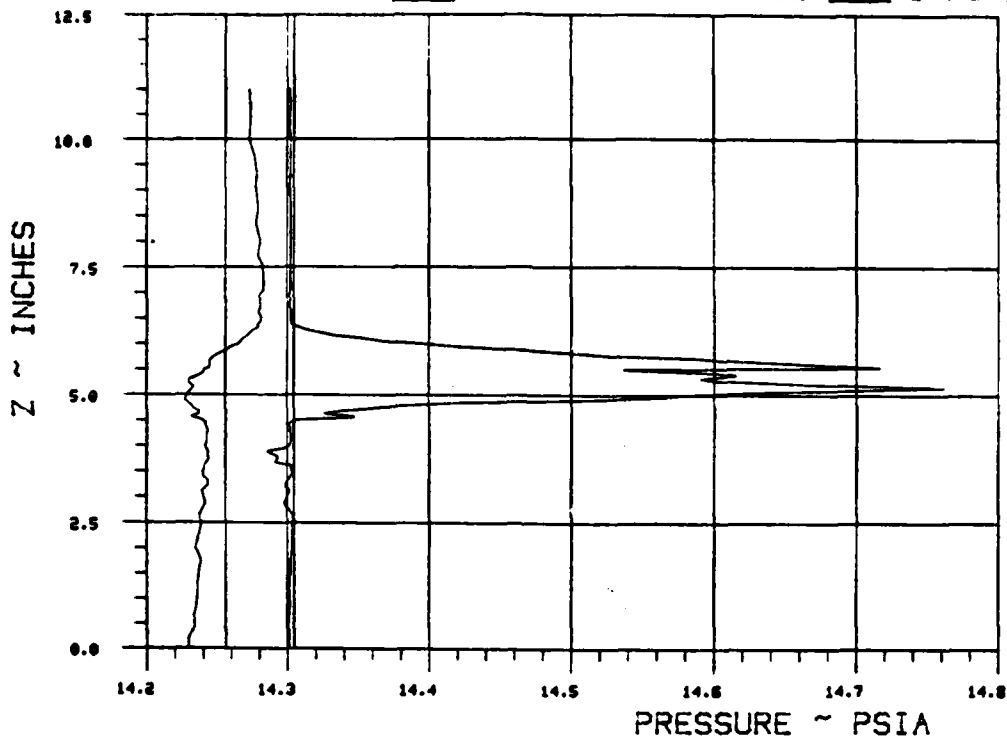


Figure II-18.

WAKE SURVEY DATA

TEST 272 RUN 24

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 50 C MU 2.0403



$X_{ss}/chord = .2419$ $Y_{ss}/chord = .0402$

BETA = 30

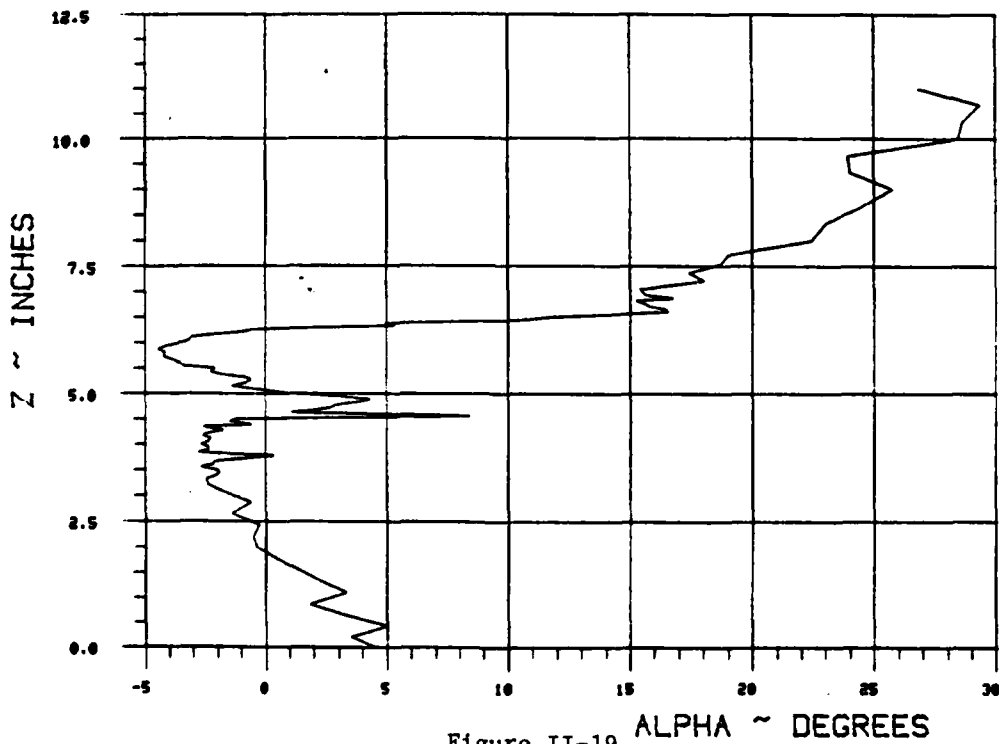
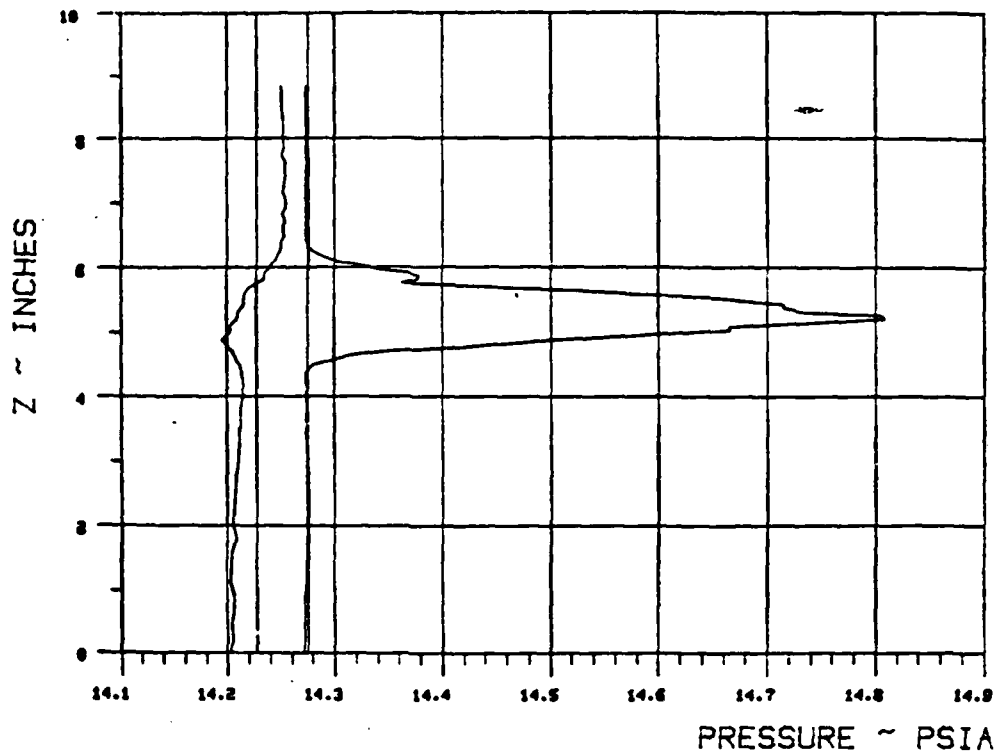


Figure II-19.

WAKE SURVEY DATA

TEST 272 RUN 24

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 50 C MU 2.0576



$X_{ss}/\text{chord} = .2411$ $Y_{ss}/\text{chord} = .0389$

BETA = 30

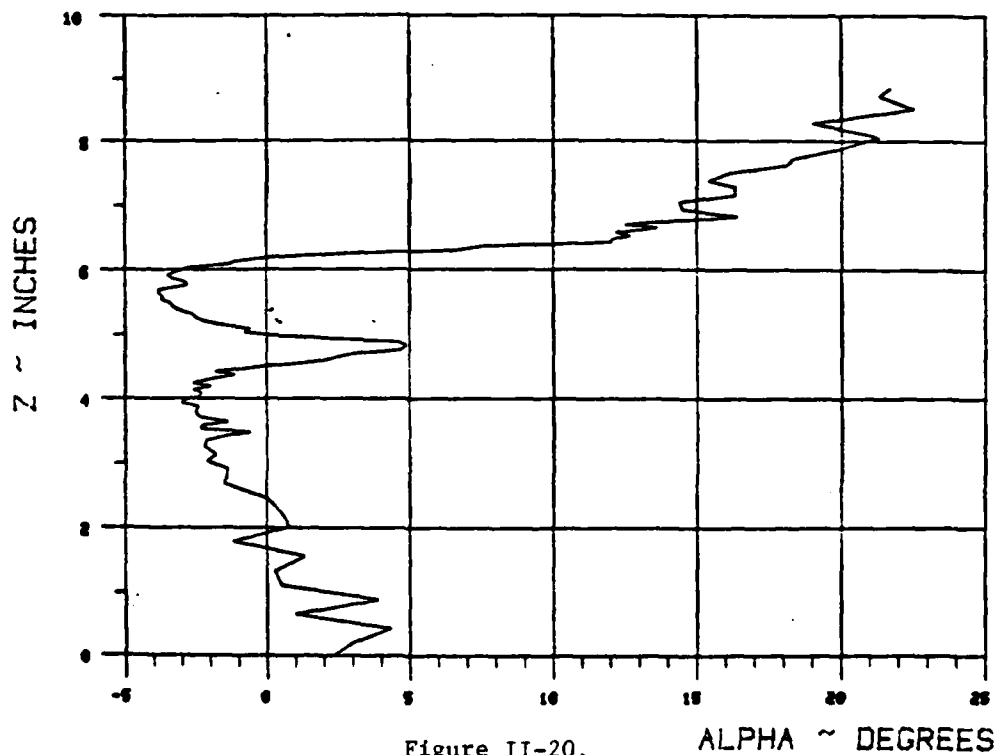
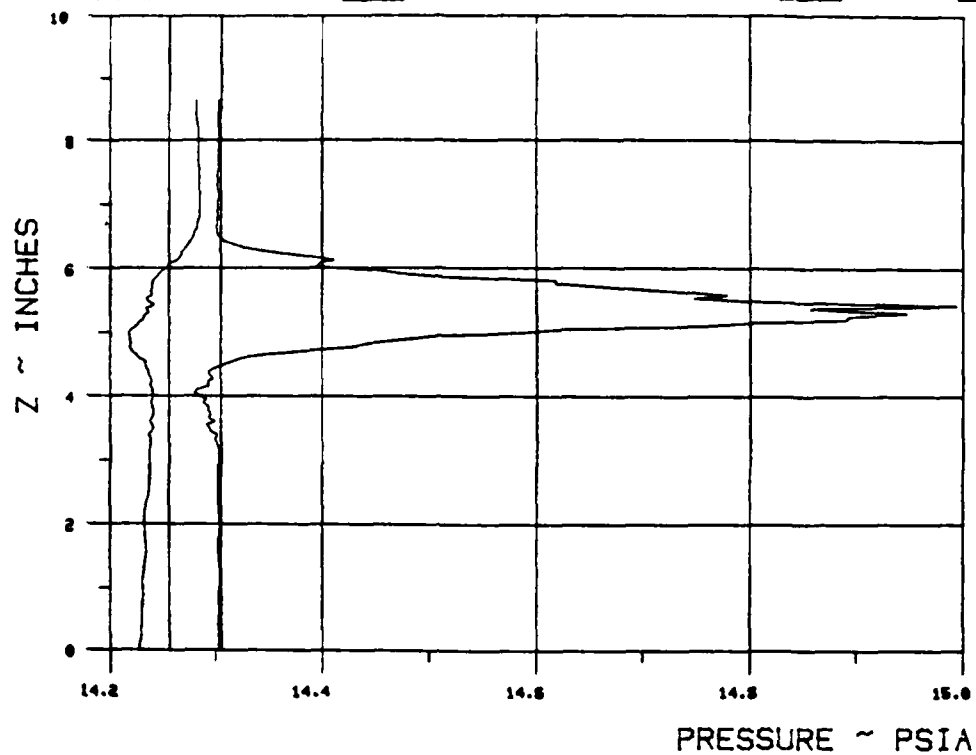


Figure II-20.

WAKE SURVEY DATA

TEST 272 RUN 28

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 50 C MU 3.0647



$X_{ss}/\text{chord} = .2409$ $Y_{ss}/\text{chord} = .0386$

BETA = 30

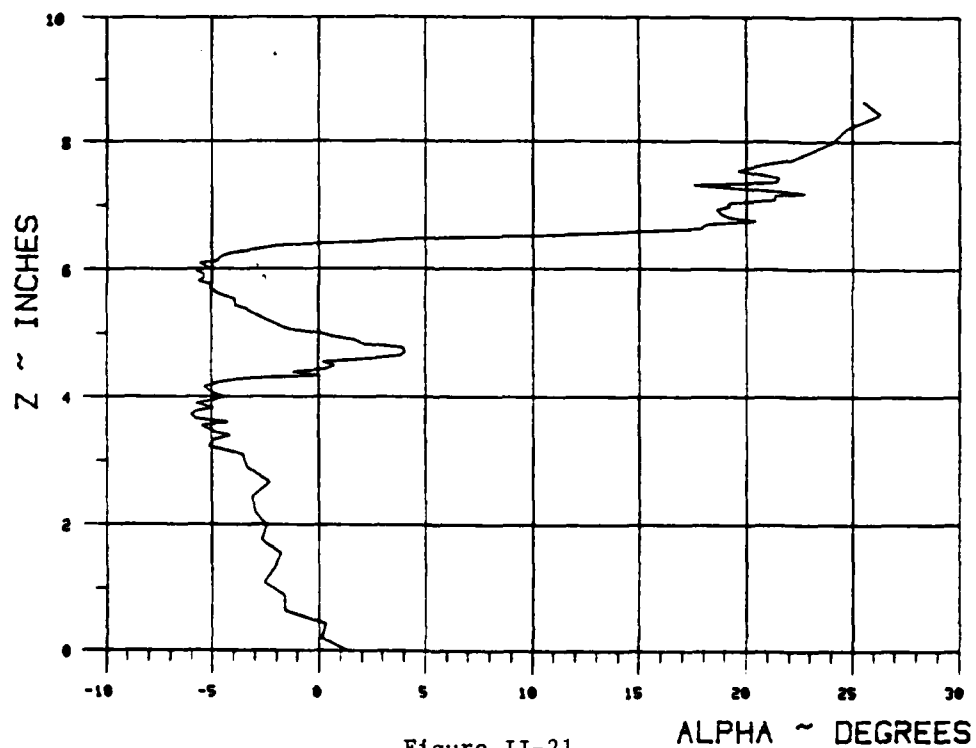
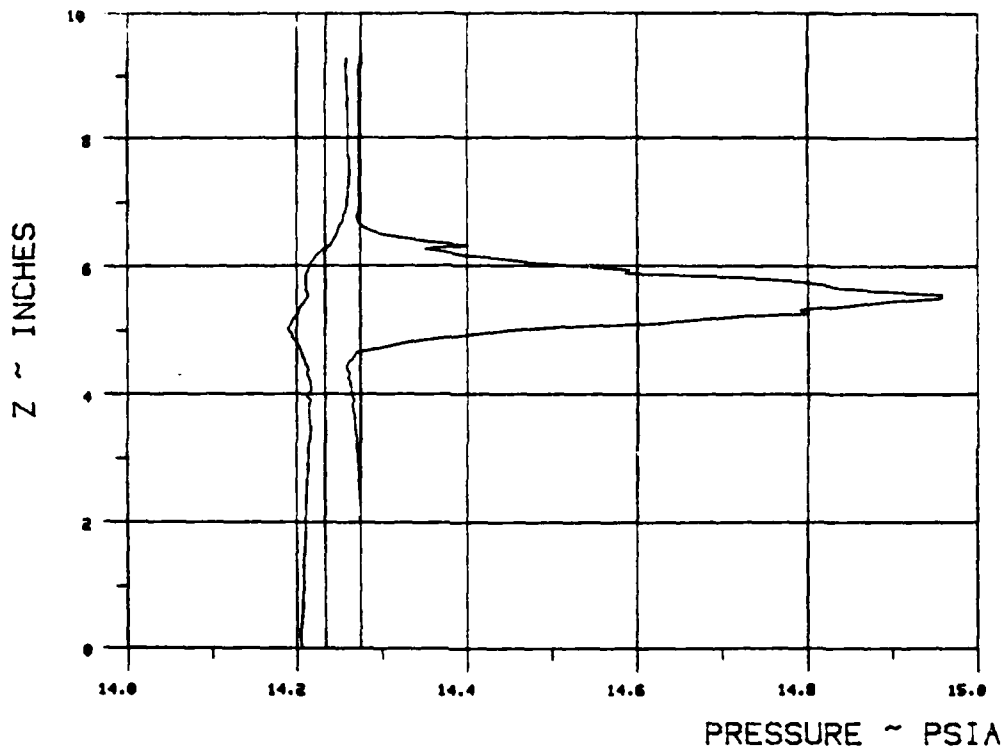


Figure II-21.

WAKE SURVEY DATA

TEST 272 RUN 22

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 60 C MU 4.1525



$X_{ss}/\text{chord} = .2413$ $Y_{ss}/\text{chord} = .0392$

BETA = 30

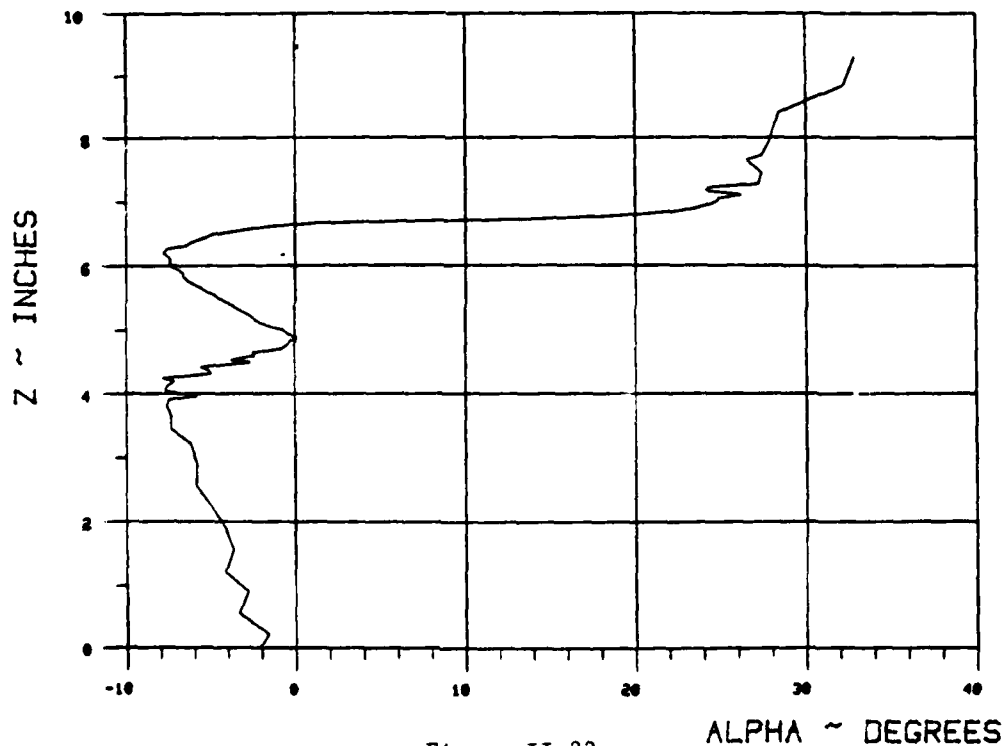
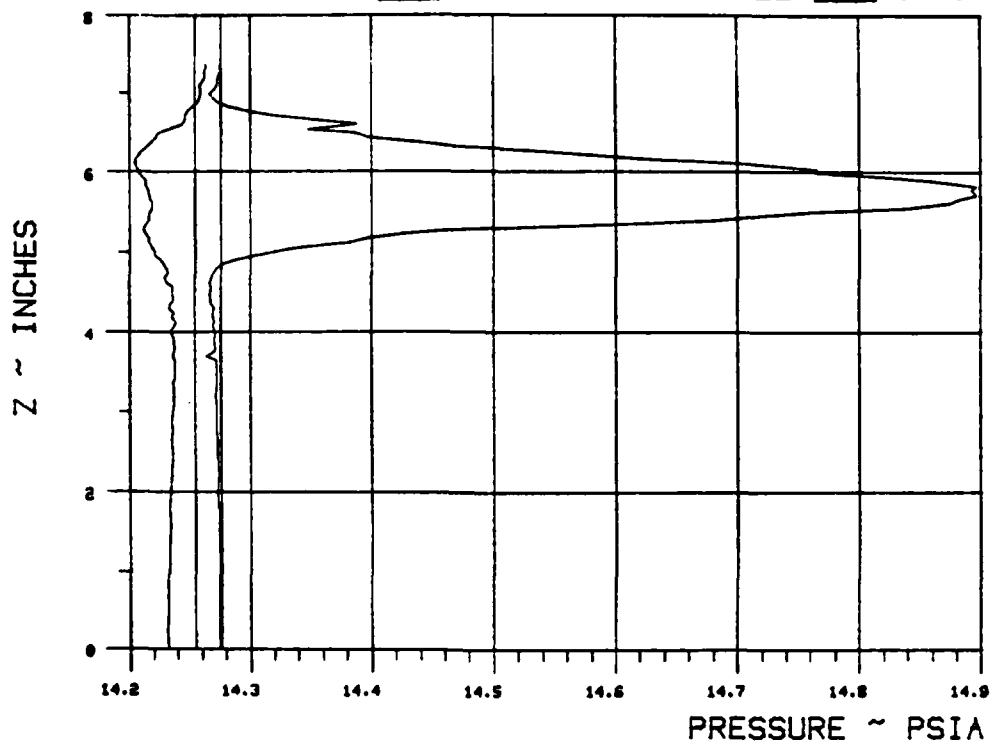


Figure II-22.

WAKE SURVEY DATA

TEST 272 RUN 25

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 50 C MU 8.1708



$X_{ee}/\text{chord} = .2406$ $Y_{ee}/\text{chord} = .0379$

BETA = 30

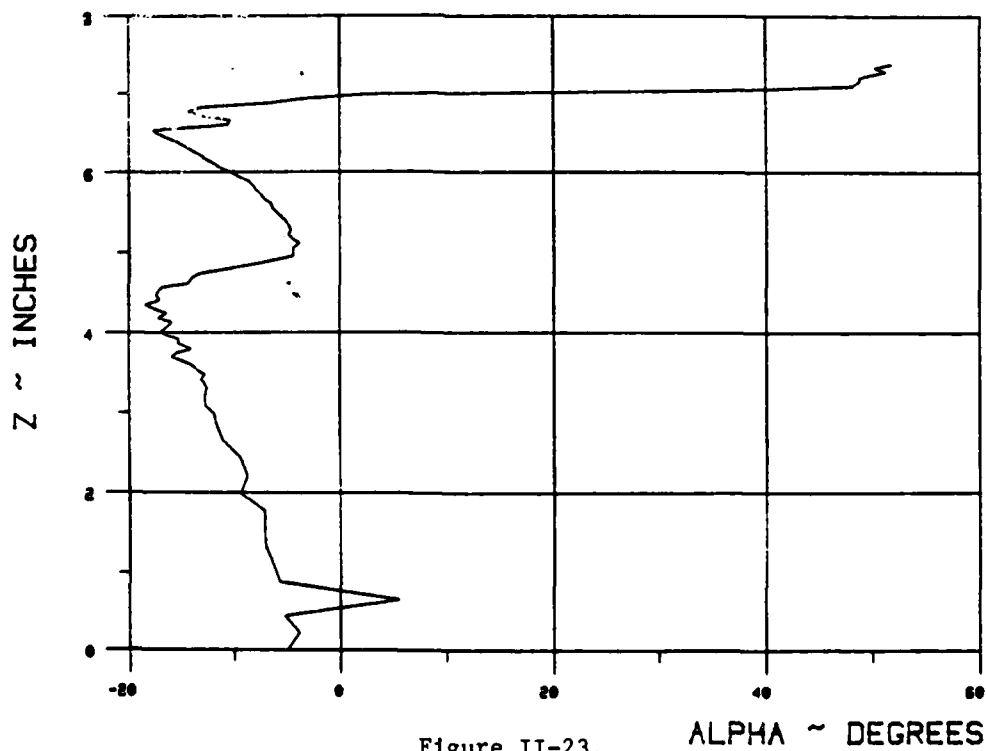
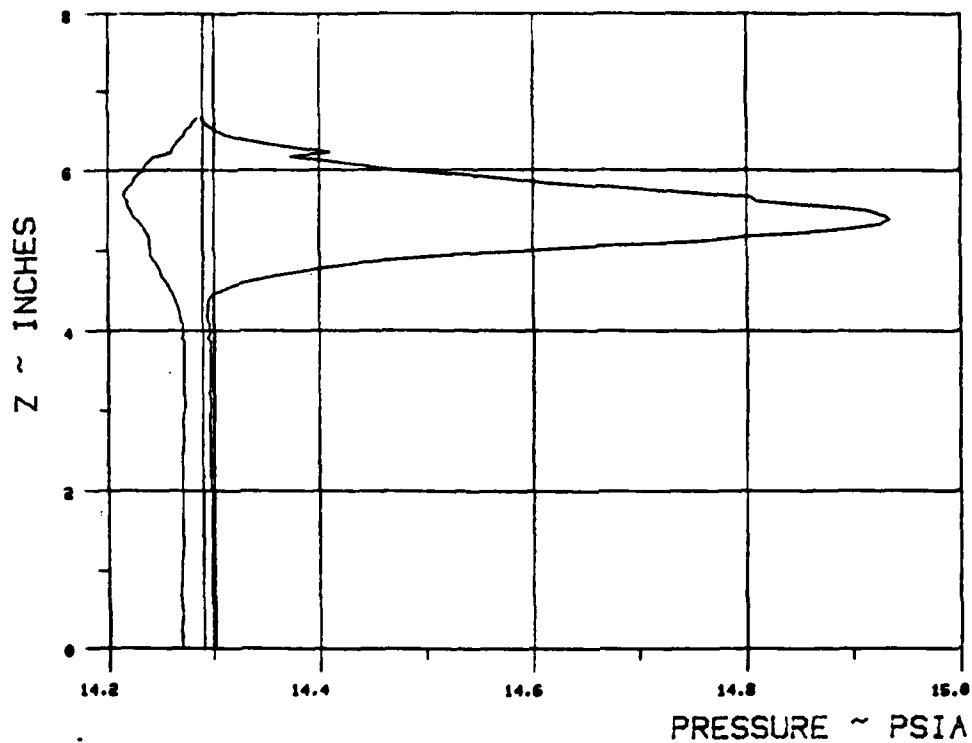


Figure II-23.

WAKE SURVEY DATA

TEST 272 RUN 26

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 50 C MU 16.2213



$X_{ee}/\text{chord} = .2305$ $Y_{ee}/\text{chord} = .0205$

BETA = 30

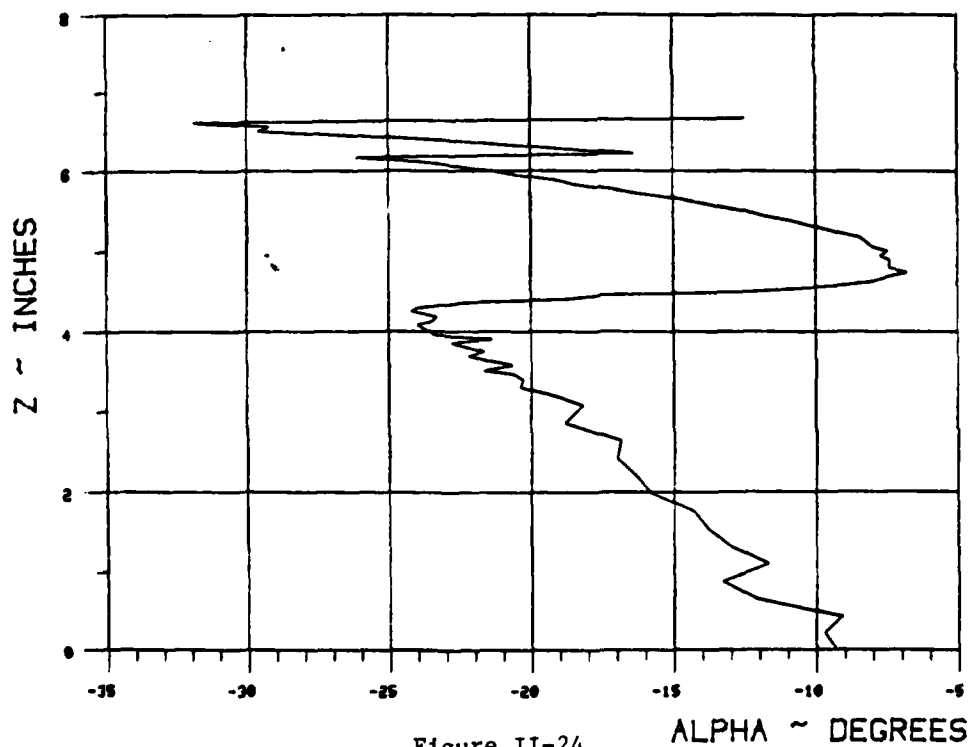
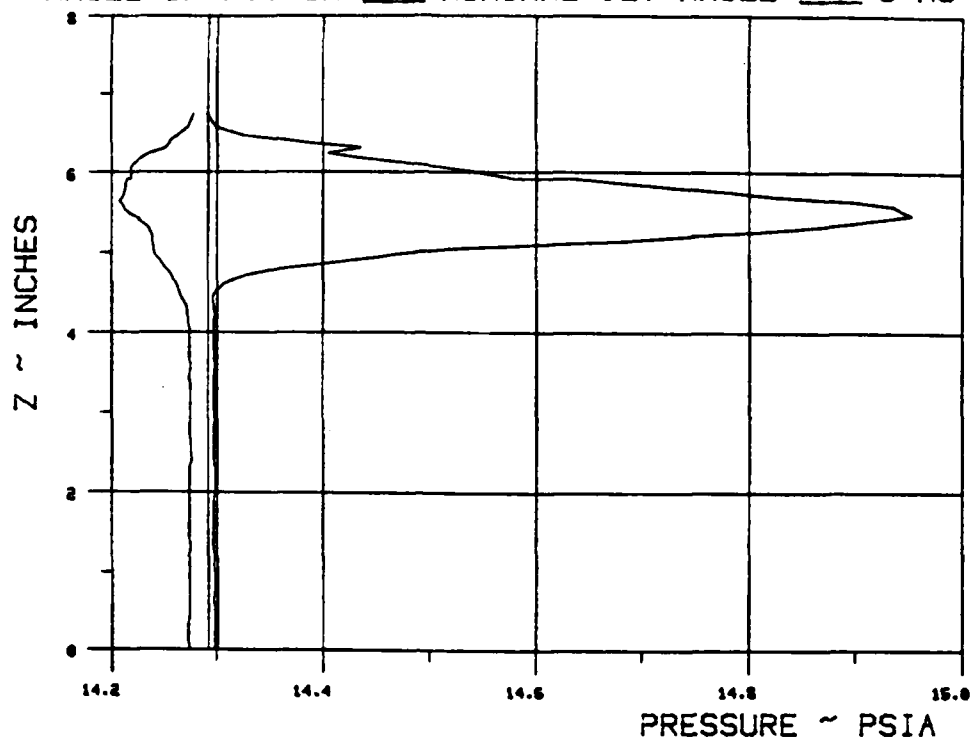


Figure II-24.

WAKE SURVEY DATA

TEST 272 RUN 27

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 50 C MU 22.3348



$X_{\text{jet}}/\text{chord} = .2309$ $Y_{\text{jet}}/\text{chord} = .0212$

BETA = 30

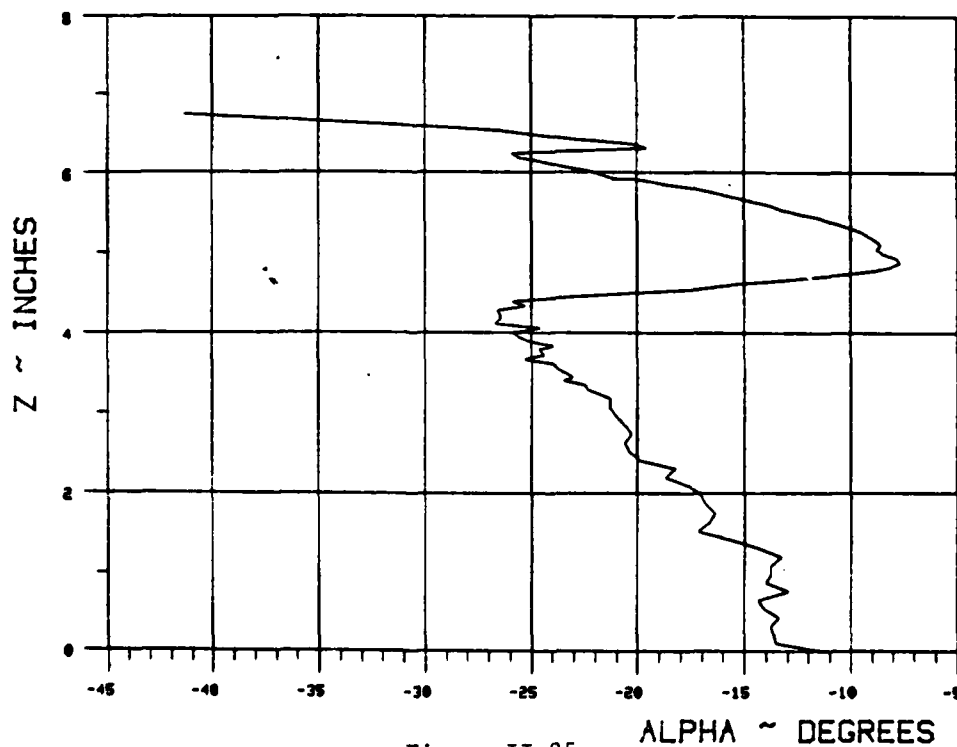
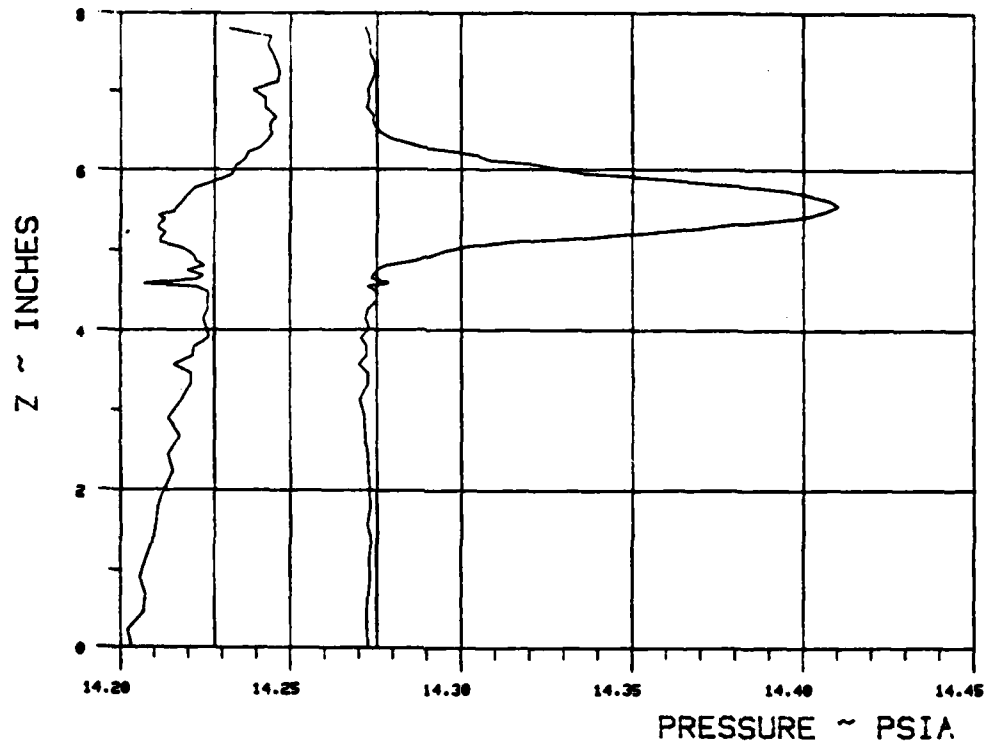


Figure II-25.

WAKE SURVEY DATA

TEST 272 RUN 30

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU .5513



$X_{ss}/\text{chord} = .1615$ $Y_{ss}/\text{chord} = -.0038$

BETA = 45.2

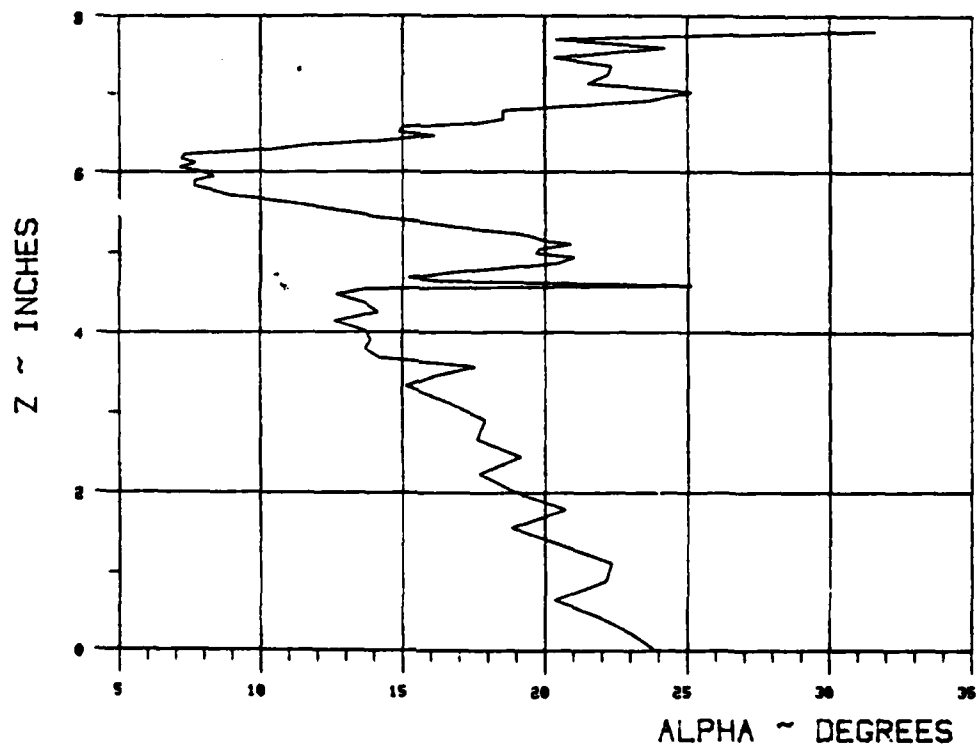
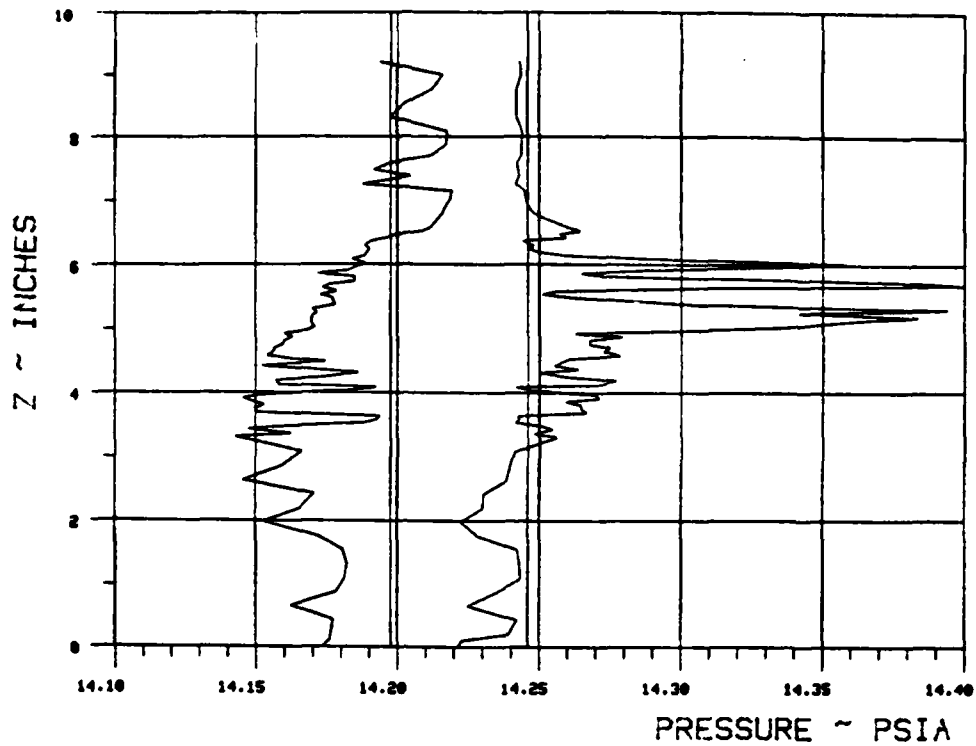


Figure II-26.

WAKE SURVEY DATA

TEST 272 RUN 43

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 1.0486



$X_{ss}/\text{chord} = .1926$ $Y_{ss}/\text{chord} = -.0353$

BETA = 46.2

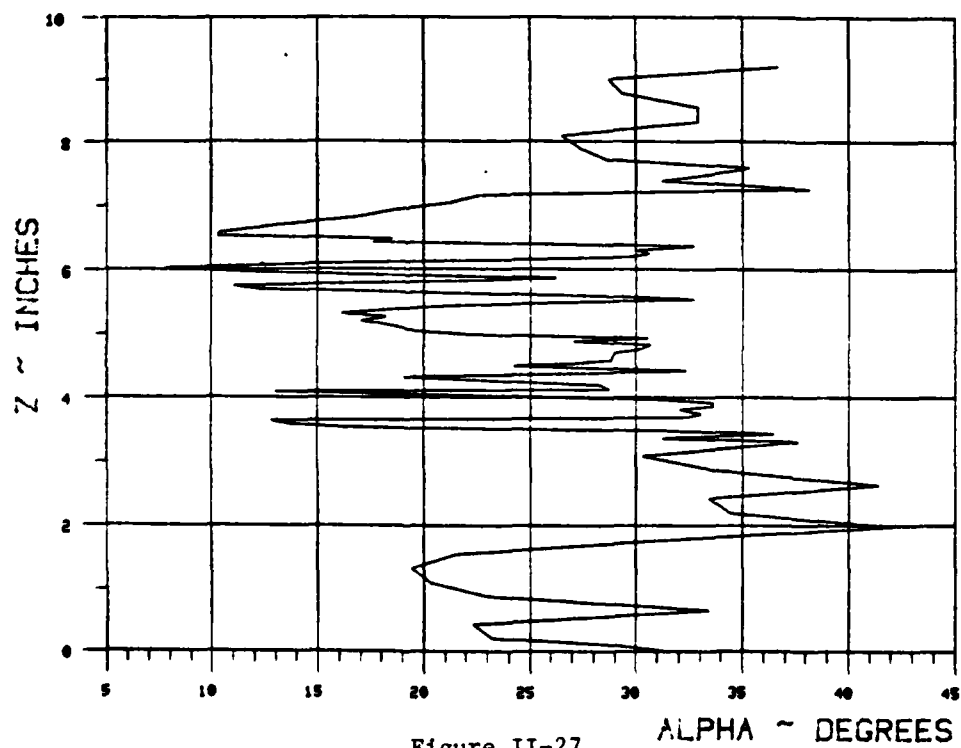
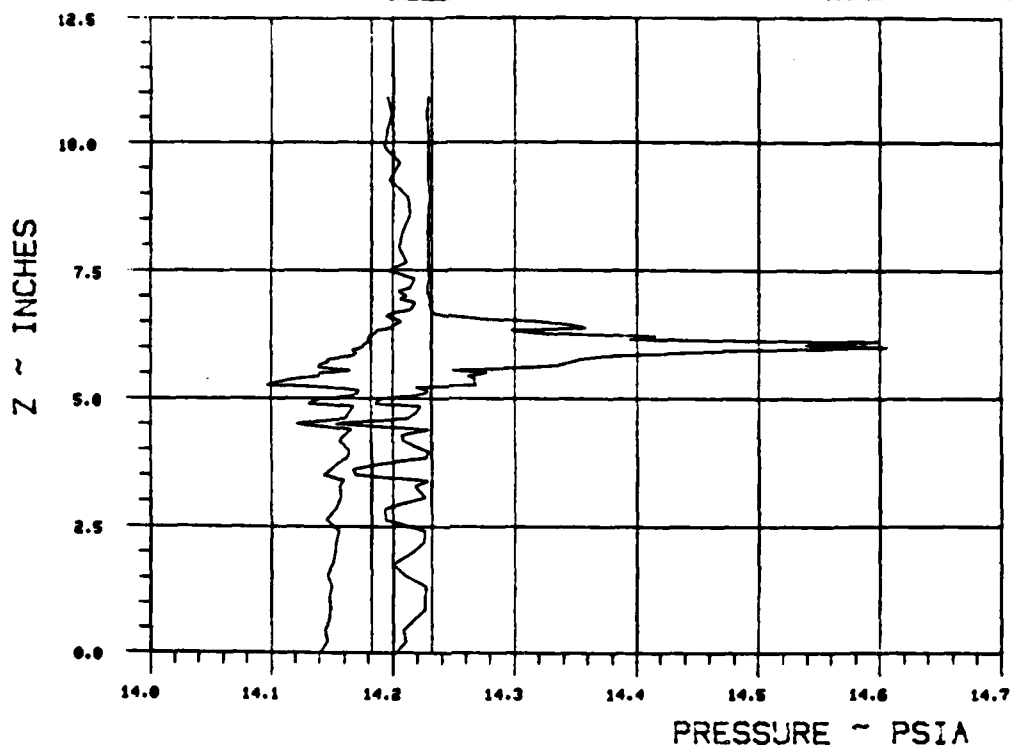


Figure II-27.

WAKE SURVEY DATA

TEST 272 RUN 41

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 1.051



$X_{ss}/\text{chord} = .1299$ $Y_{ss}/\text{chord} = .0273$

BETA = 45.2

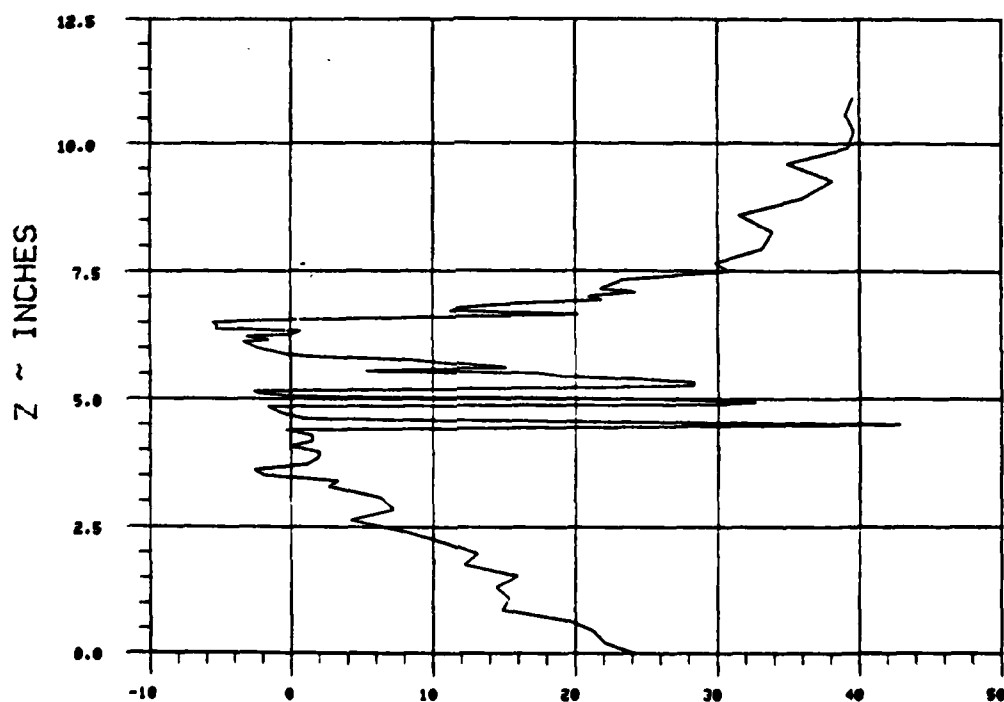
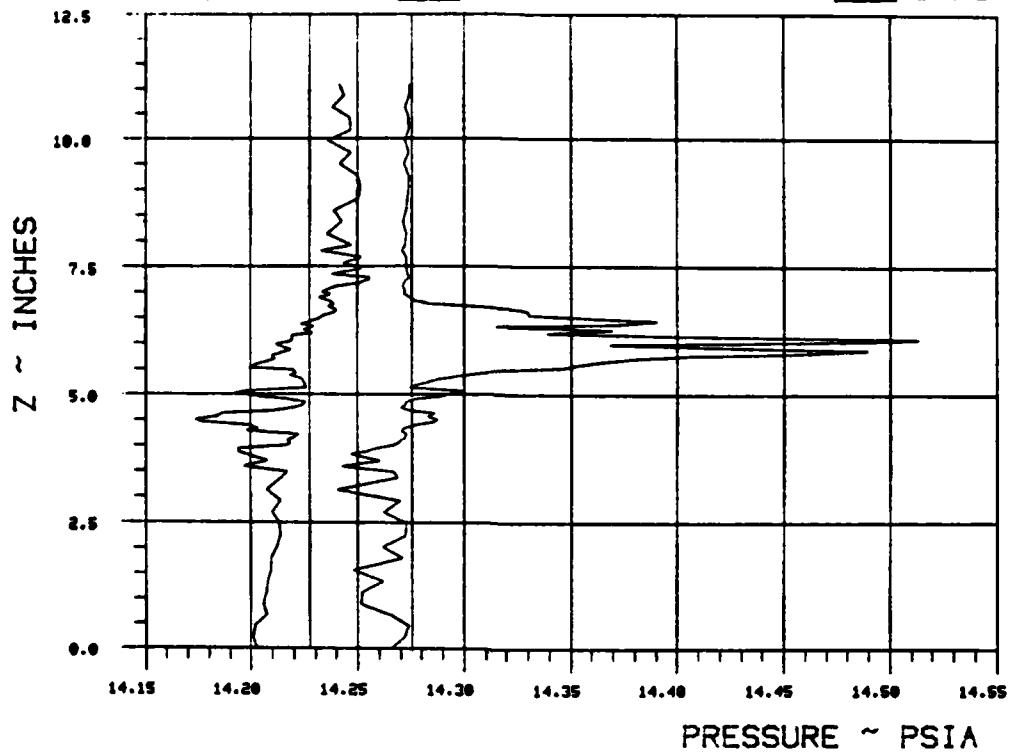


Figure II-28. ALPHA ~ DEGREES

WAKE SURVEY DATA

TEST 272 RUN 31

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 1.0553



$X_{ss}/\text{chord} = \underline{.1612}$ $Y_{ss}/\text{chord} = \underline{-.004}$

BETA = 45.2

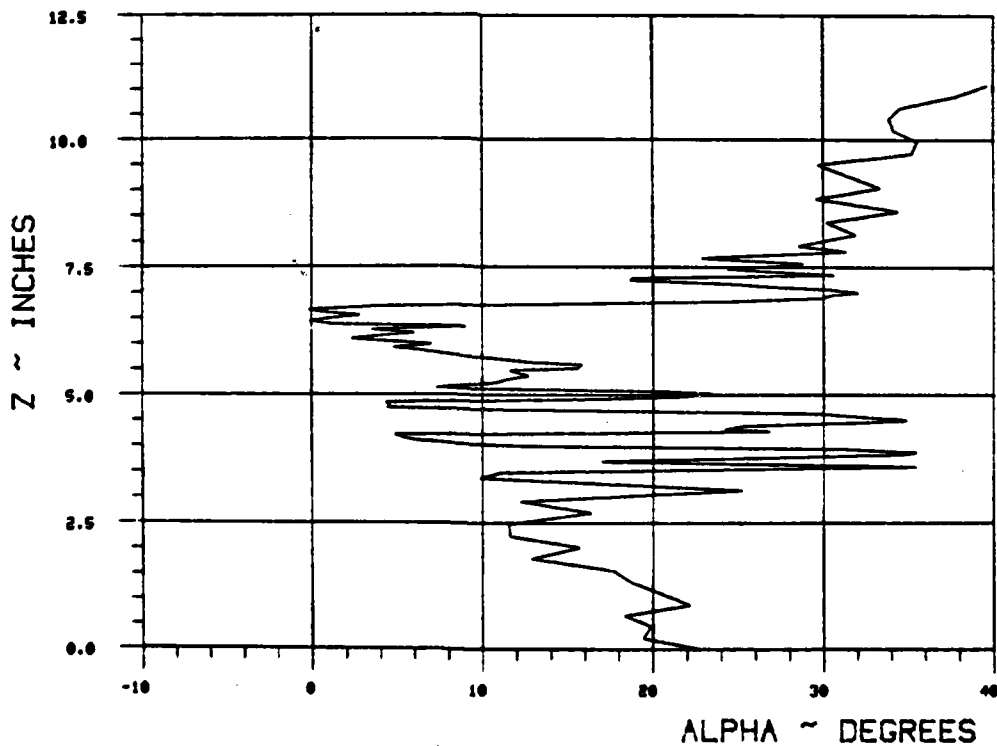
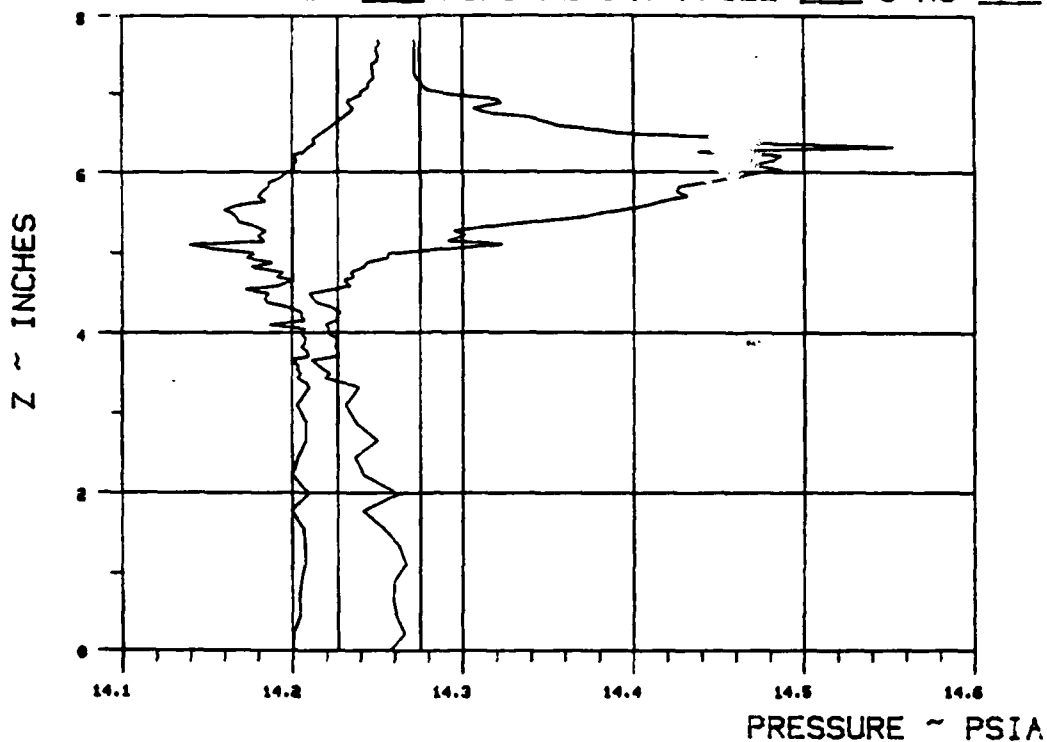


Figure II-29.

WAKE SURVEY DATA

TEST 272 RUN 32

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 2.0838



$X_{ss}/\text{chord} = .1619$ $Y_{ss}/\text{chord} = -.0033$

BETA = 45.2

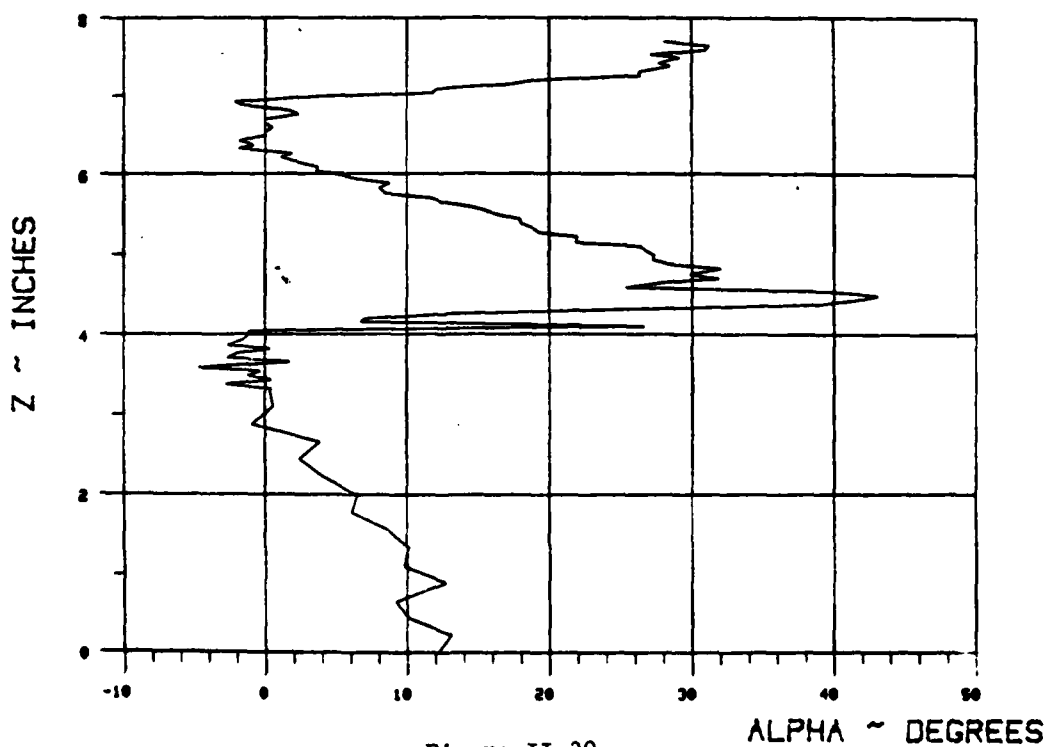
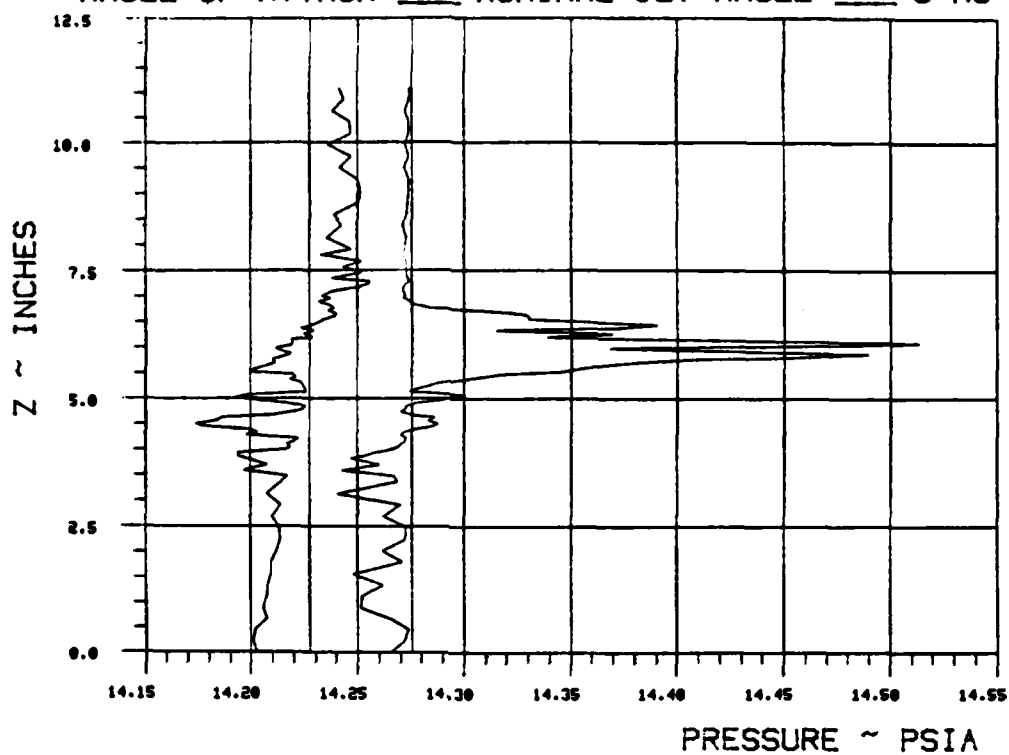


Figure II-30.

WAKE SURVEY DATA

TEST 272 RUN 31

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 7.0 C MU 1.0553



$X_{ee}/\text{chord} = .1612$ $Y_{ee}/\text{chord} = -.004$

BETA = 45.2

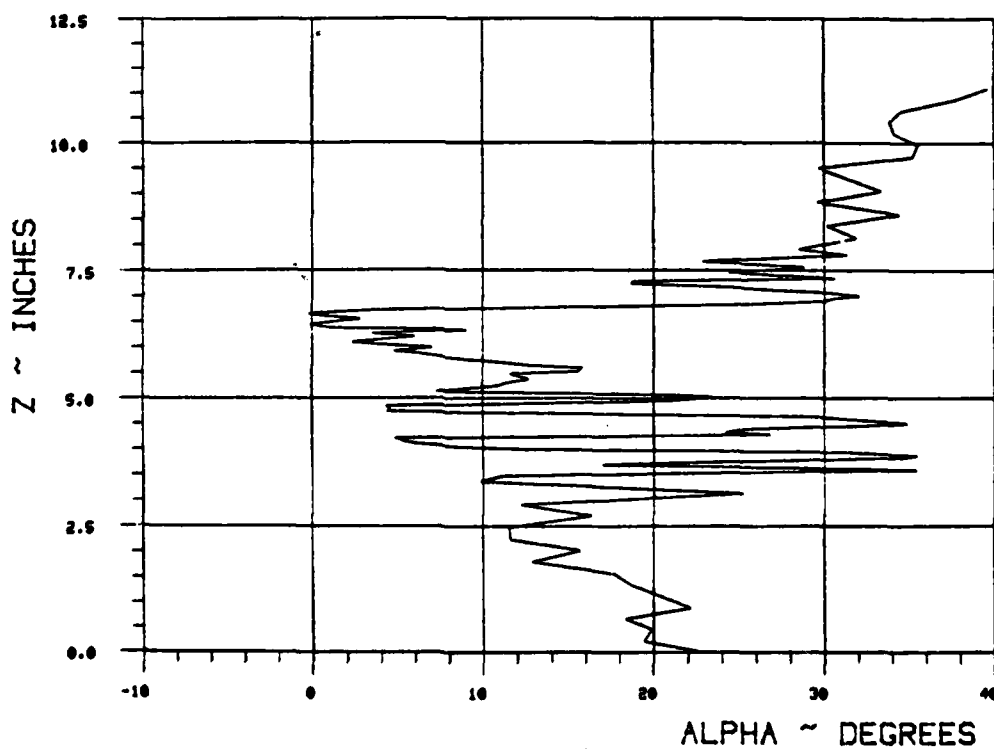
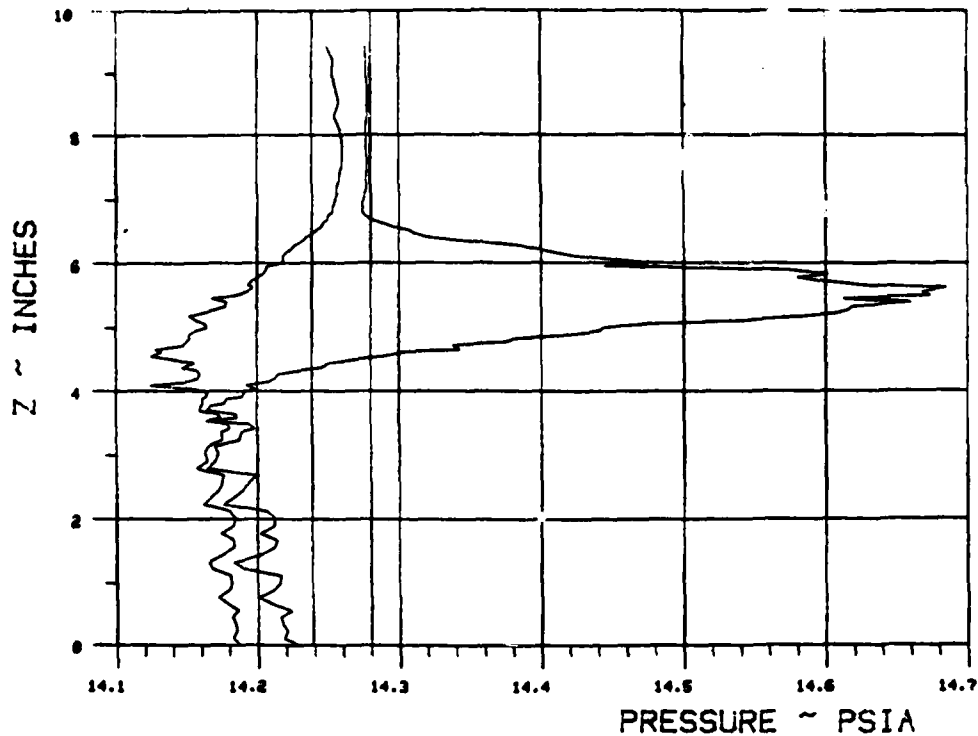


Figure II-29.

WAKE SURVEY DATA

TEST 272 RUN 36

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 4.2603



$X_{ee}/\text{chord} = .1468$ $Y_{ee}/\text{chord} = -.0185$

BETA = 45.2

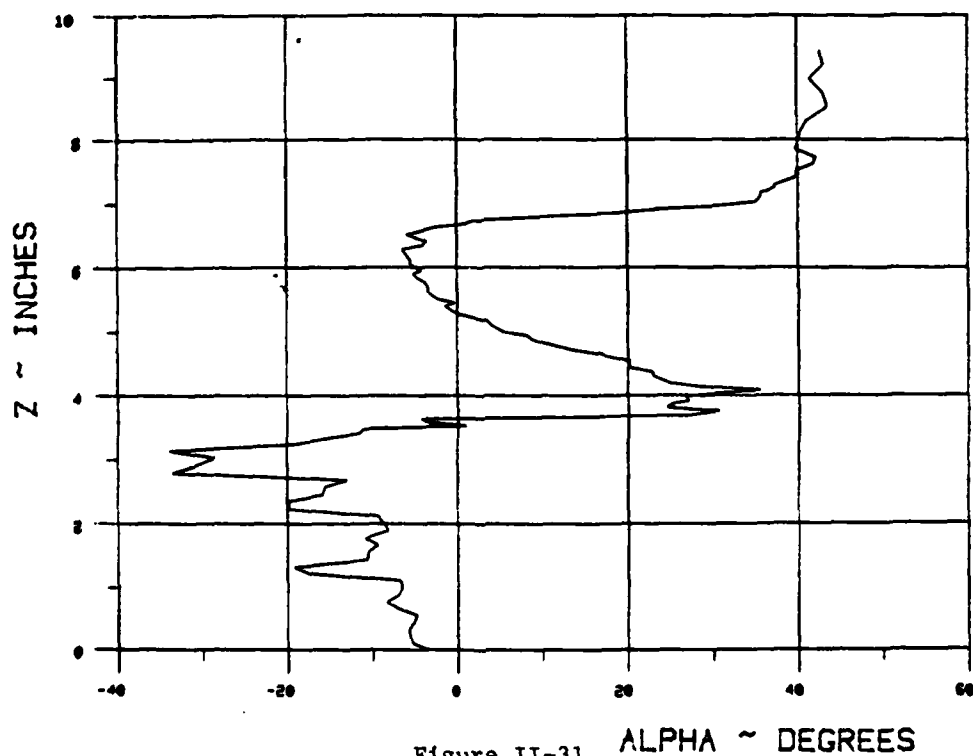
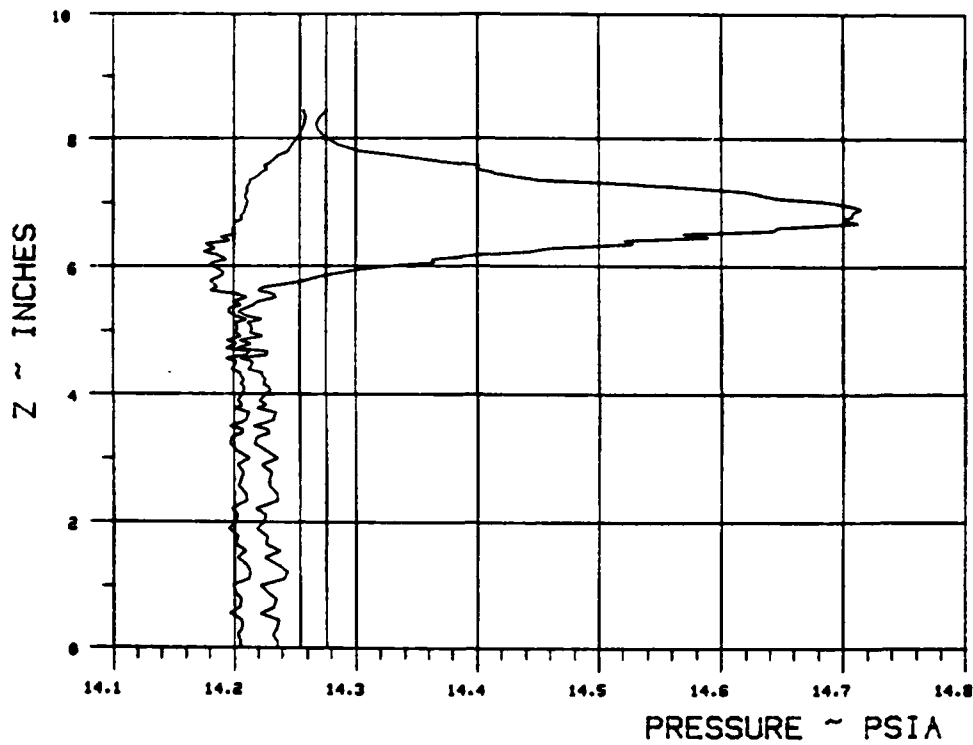


Figure II-31. ALPHA ~ DEGREES

WAKE SURVEY DATA

TEST 272 RUN 34

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 8.319



$X_{ss}/\text{chord} = .1622$ $Y_{ss}/\text{chord} = -.0031$

BETA = 45.2

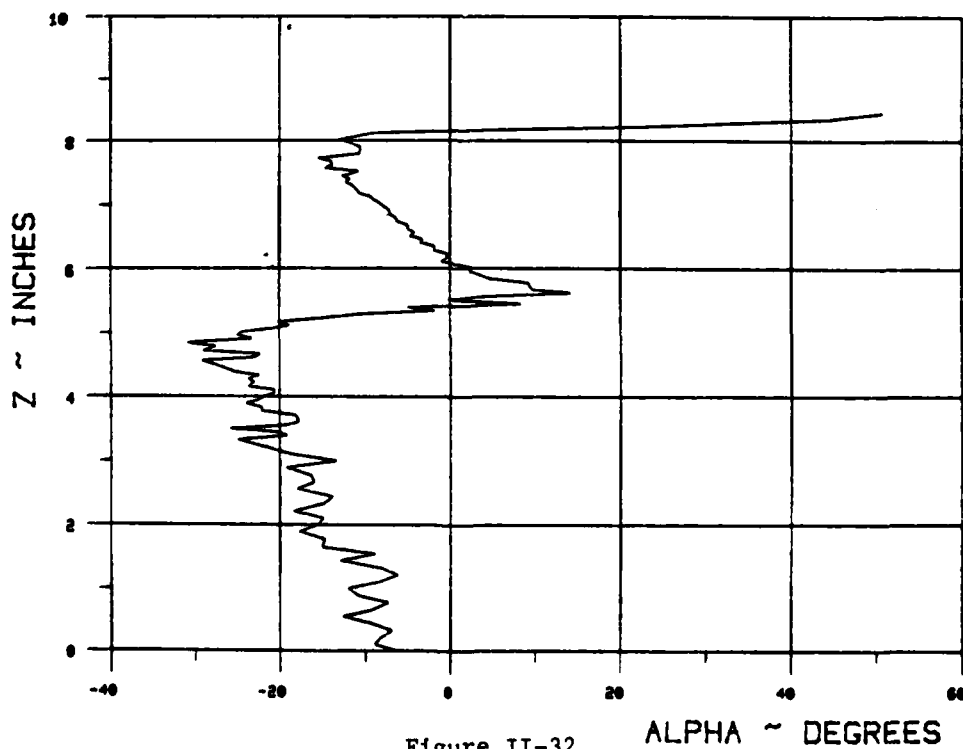
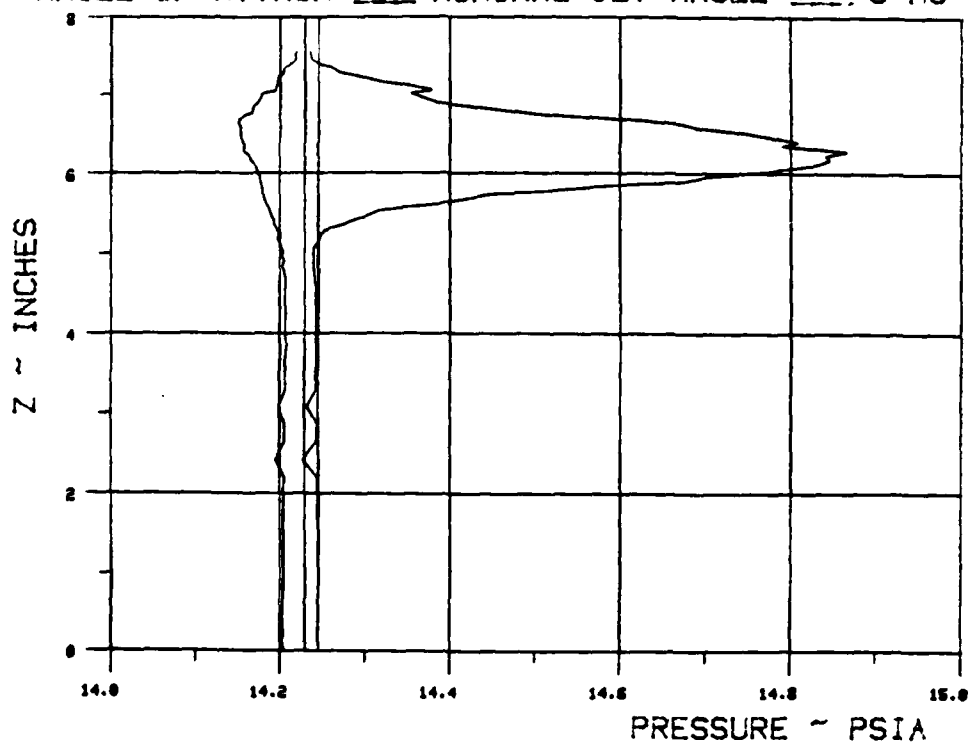


Figure II-32.

WAKE SURVEY DATA

TEST 272 RUN 42

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 10.524



$X_{ee}/\text{chord} = .1472$ $Y_{ee}/\text{chord} = -.018$

BETA = 45.2

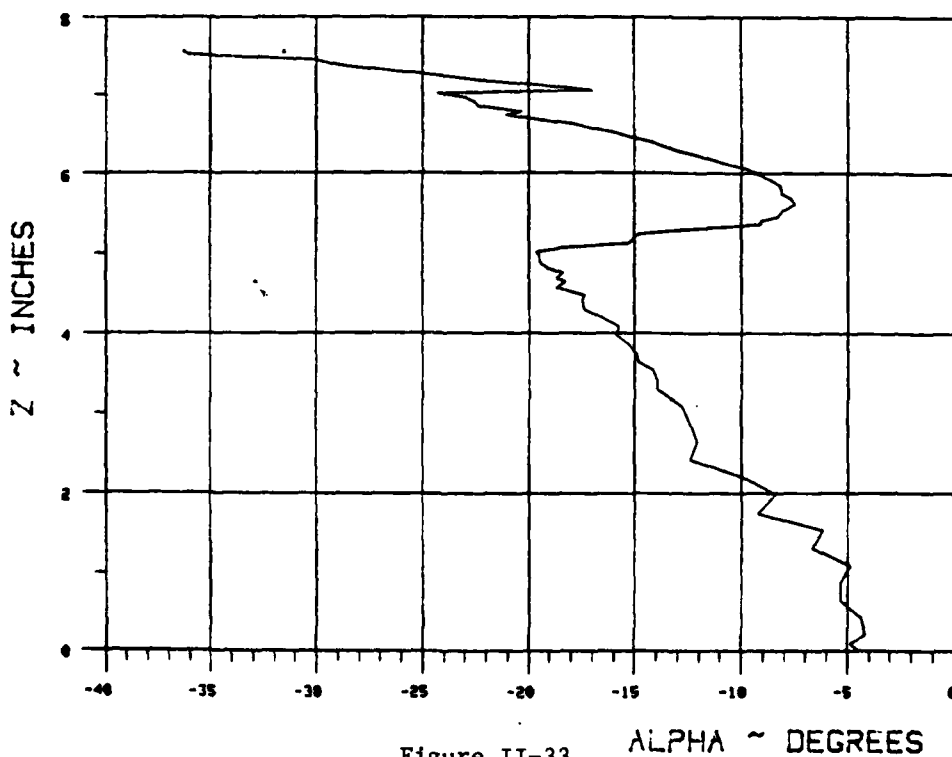
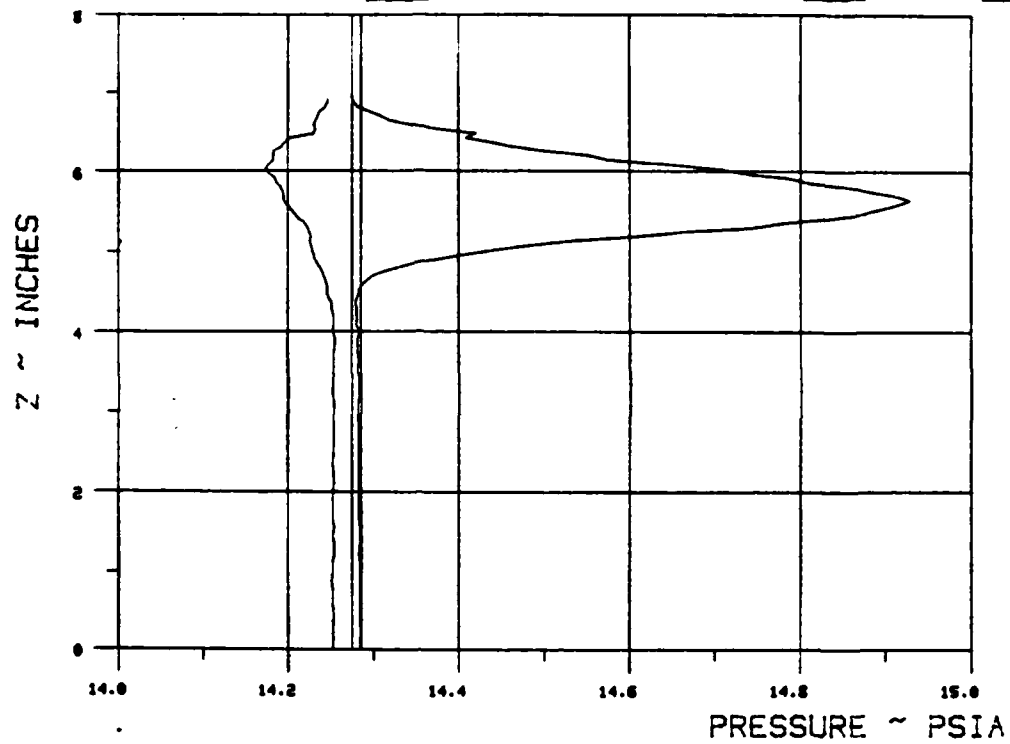


Figure II-33. ALPHA ~ DEGREES

WAKE SURVEY DATA

TEST 272 RUN 37

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 16.6099



$X_{ss}/\text{chord} = .136$ $Y_{ss}/\text{chord} = -.0291$

BETA = 45.2

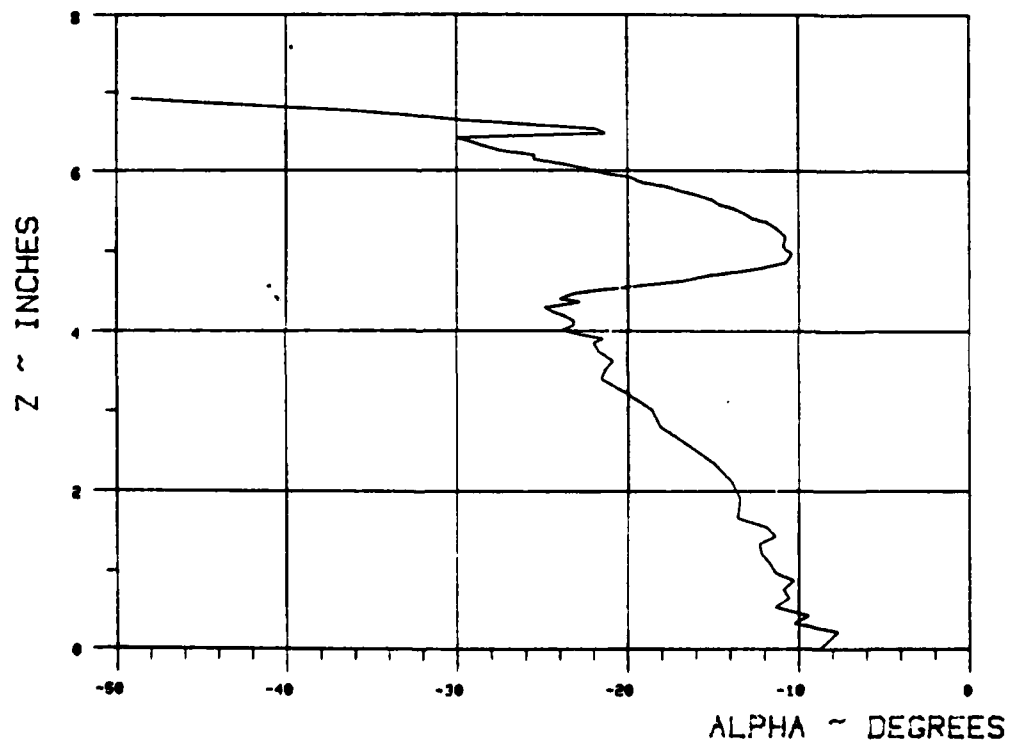
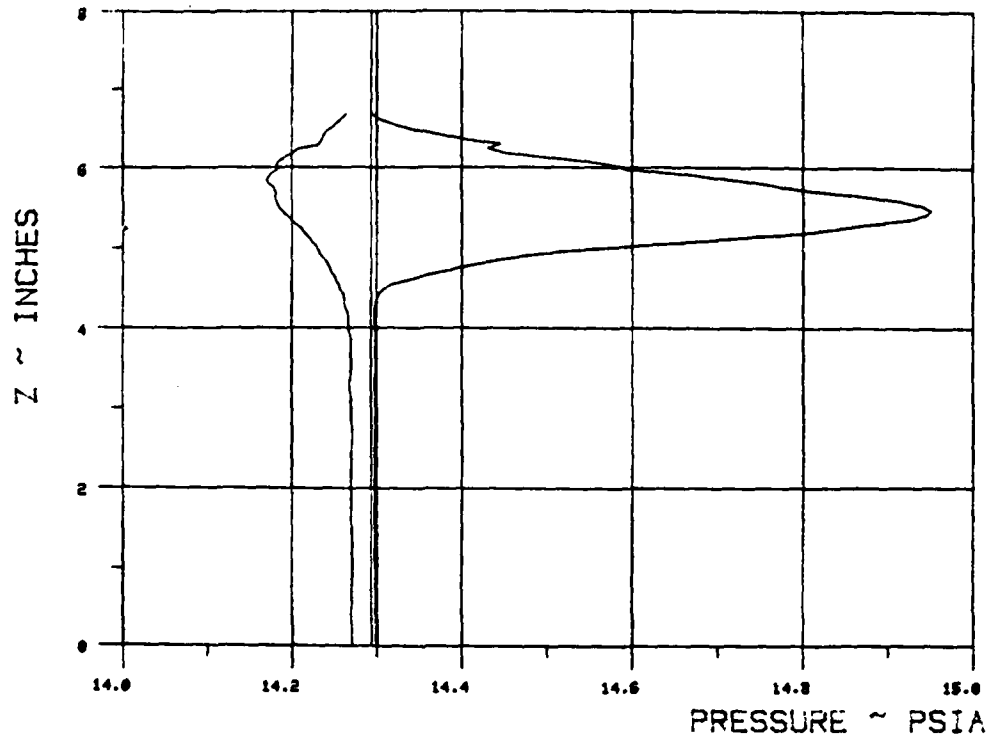


Figure II-34.

WAKE SURVEY DATA

TEST 272 RUN 30

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 24.4346



$X_{ss}/\text{chord} = .1322$ $Y_{ss}/\text{chord} = -.0329$

BETA = 45.2

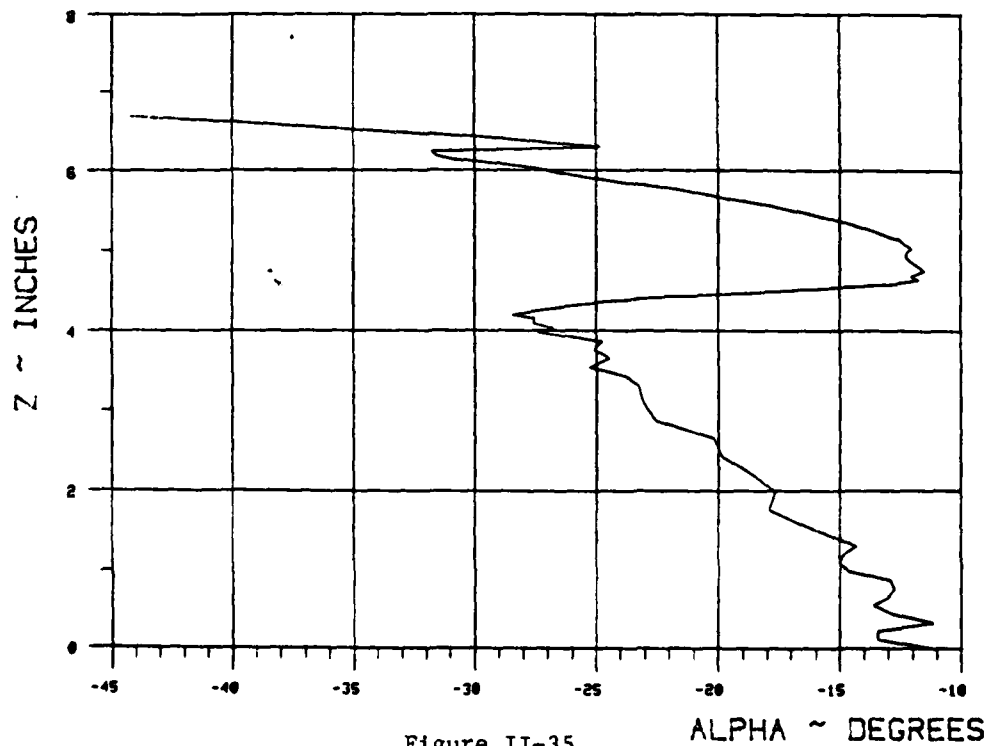
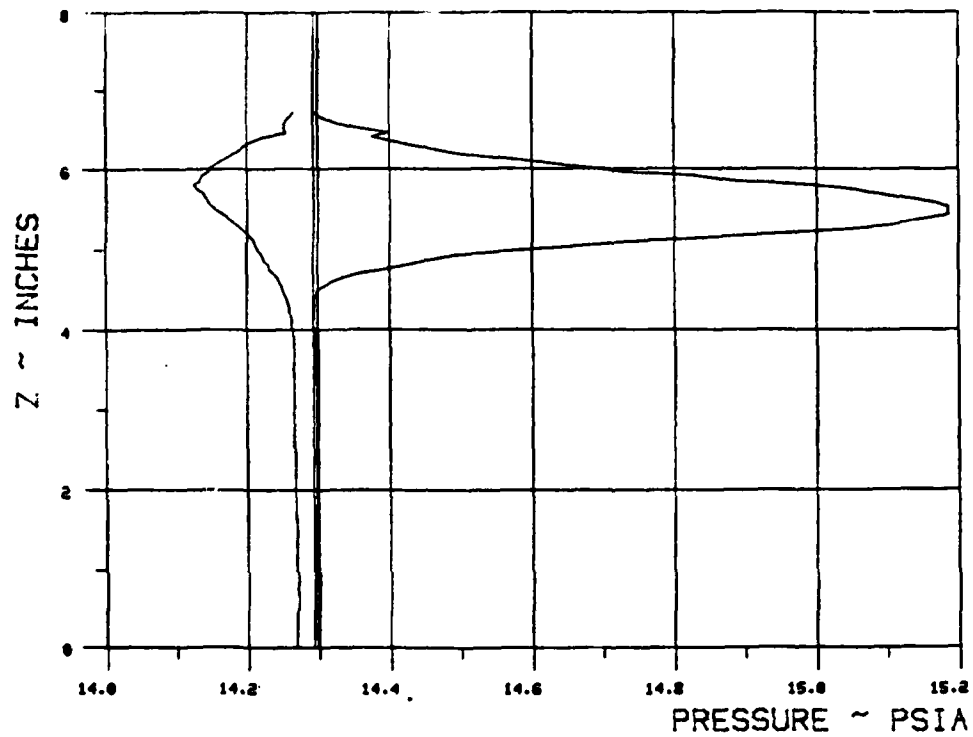


Figure II-35.

WAKE SURVEY DATA

TEST 272 RUN 39

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 28.9276



$X_{ss}/\text{chord} = .132$ $Y_{ss}/\text{chord} = .033/$

BETA = 45.2

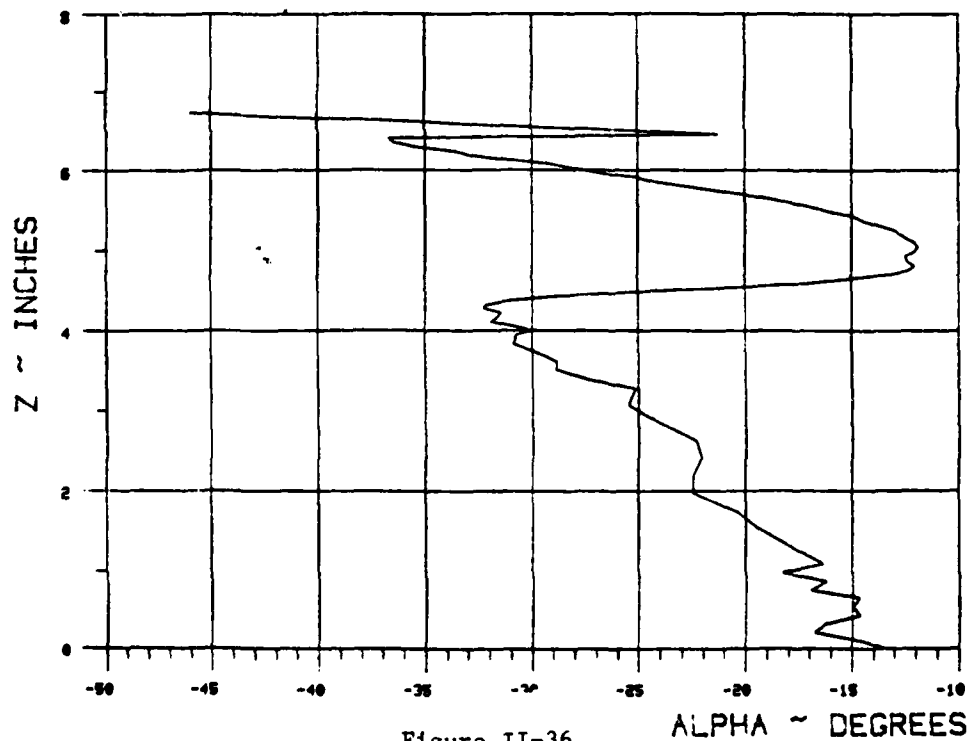
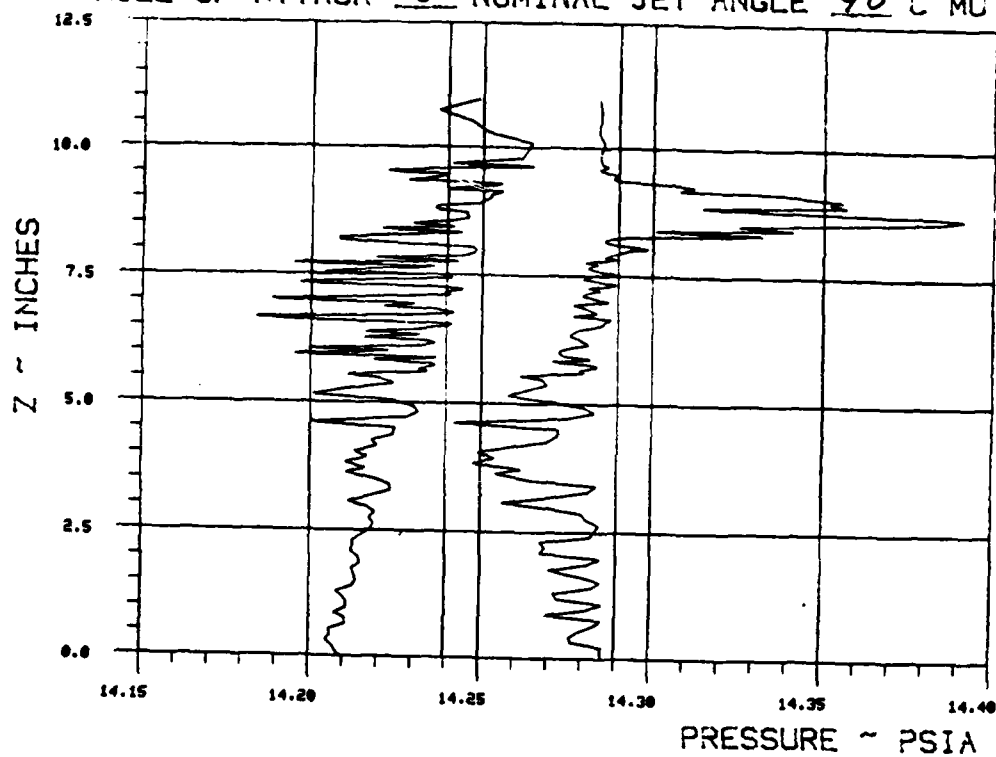


Figure II-36.

WAKE SURVEY DATA

TEST 272 RUN 44

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU .5101



$X_{es}/chord = .1922$ $Y_{es}/chord = .0267$

BETA = 45.2

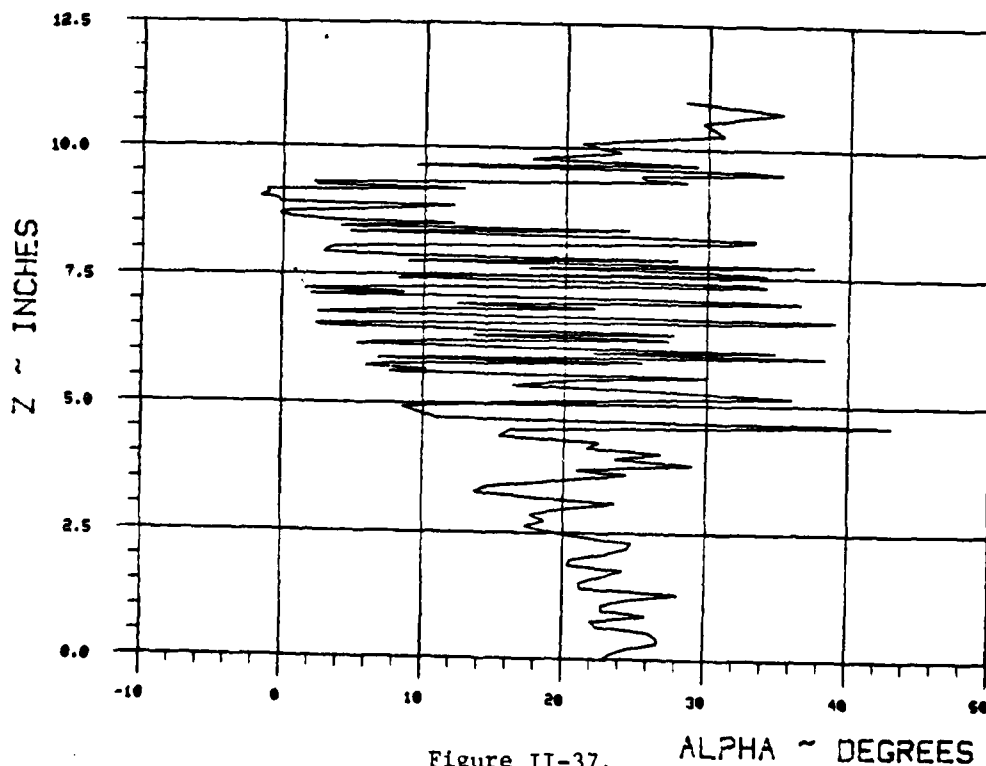
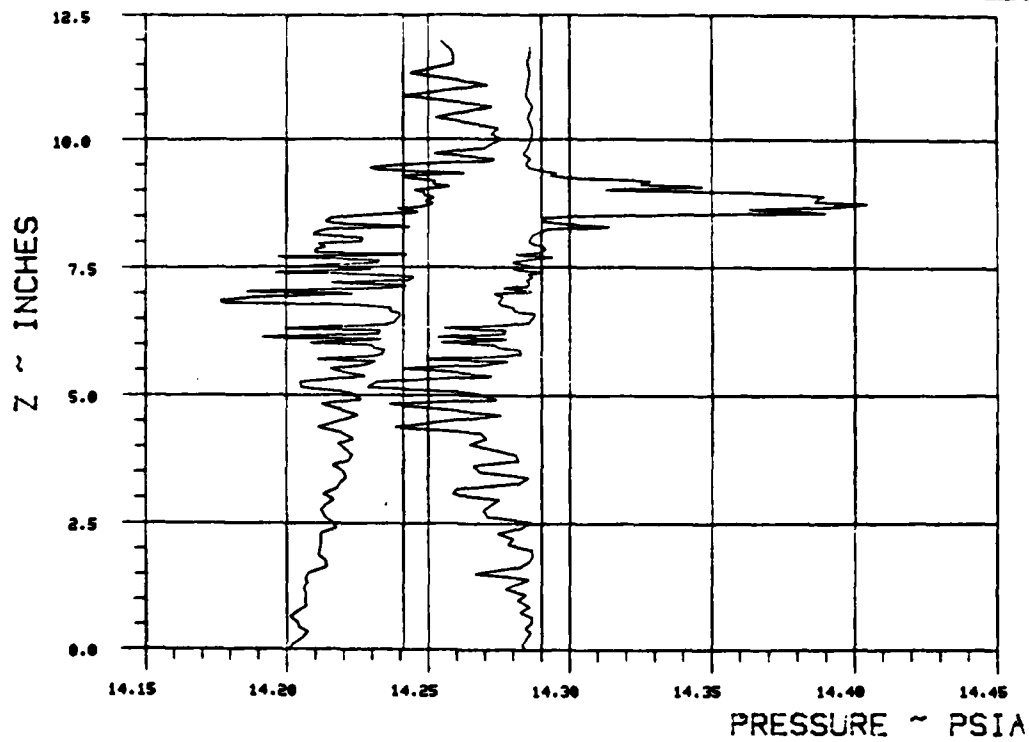


Figure II-37.

WAKE SURVEY DATA

TEST 272 RUN 47

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU .5201



$X_{ss}/\text{chord} = .1759$ $Y_{ss}/\text{chord} = .0418$

BETA = 45.2

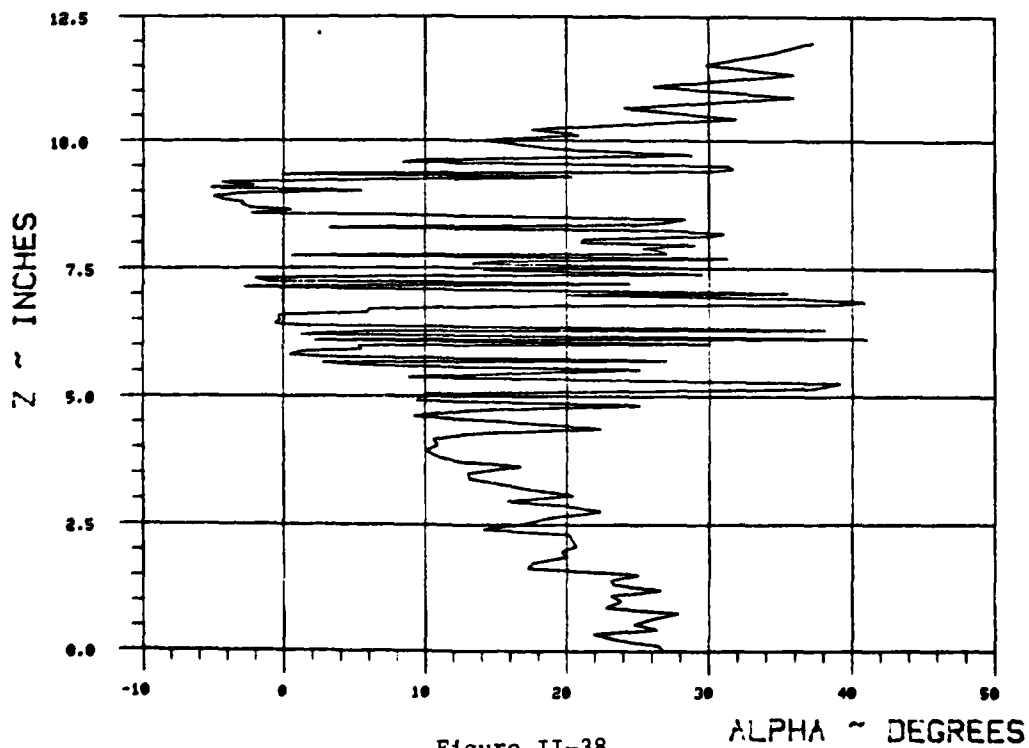
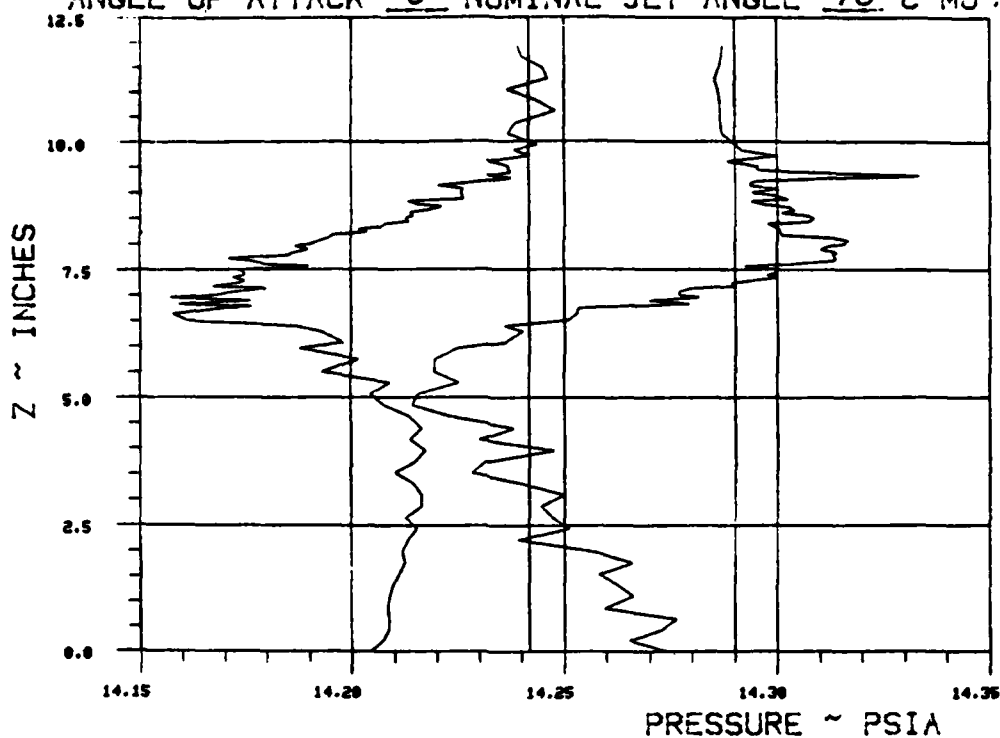


Figure II-38.

WAKE SURVEY DATA

TEST 272 RUN 45

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU .9785



$X_{ee}/\text{chord} = .1922$ $Y_{ee}/\text{chord} = .0267$

BETA = 45.2

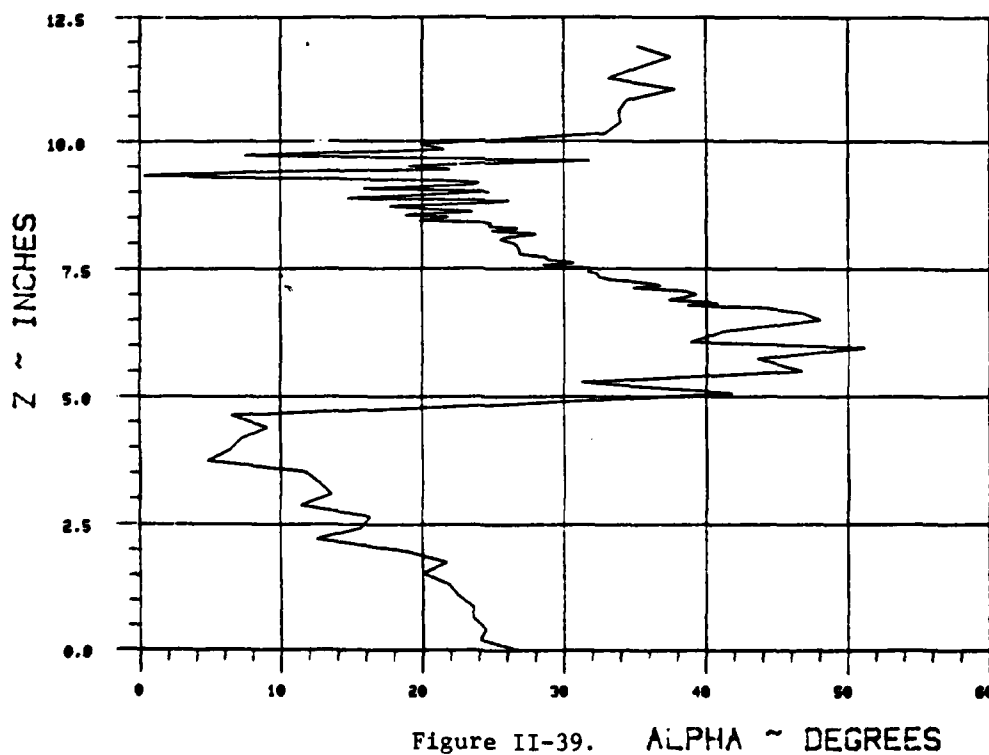
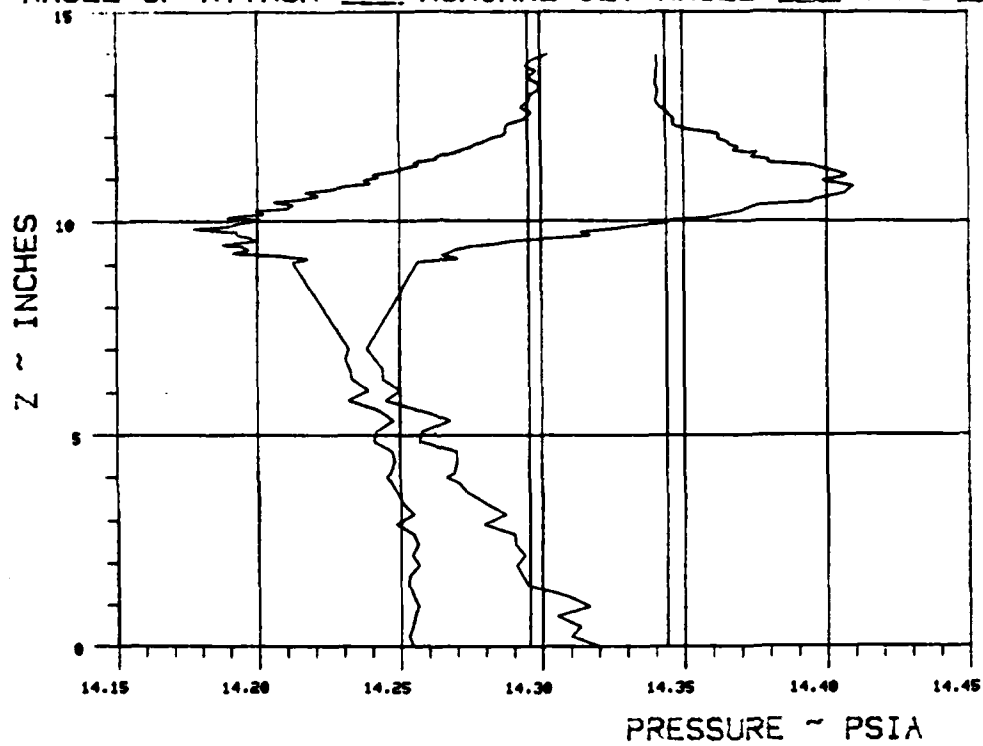


Figure II-39. ALPHA ~ DEGREES

WAKE SURVEY DATA

TEST 272 RUN 56

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 1.9504



$X_{ss}/\text{chord} = .1995$ $Y_{ss}/\text{chord} = .0444$

BETA = 55

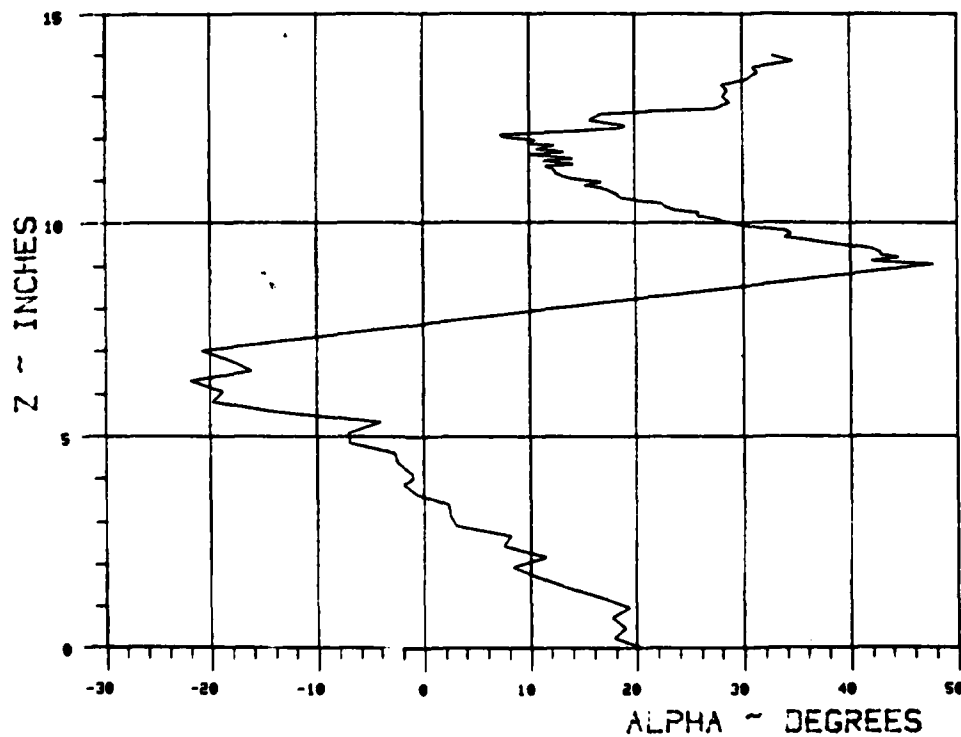
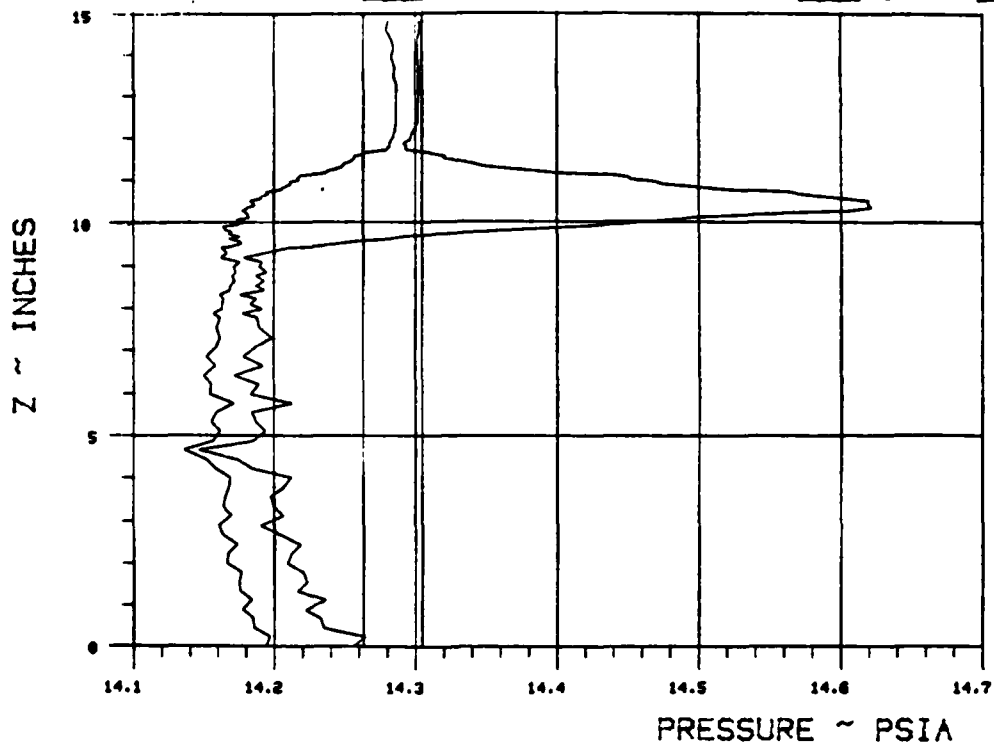


Figure II-40.

WAKE SURVEY DATA

TEST 272 RUN 48

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 3.9373



$X_{se}/\text{chord} = .191$ $Y_{se}/\text{chord} = .0568$

BETA = 45.2

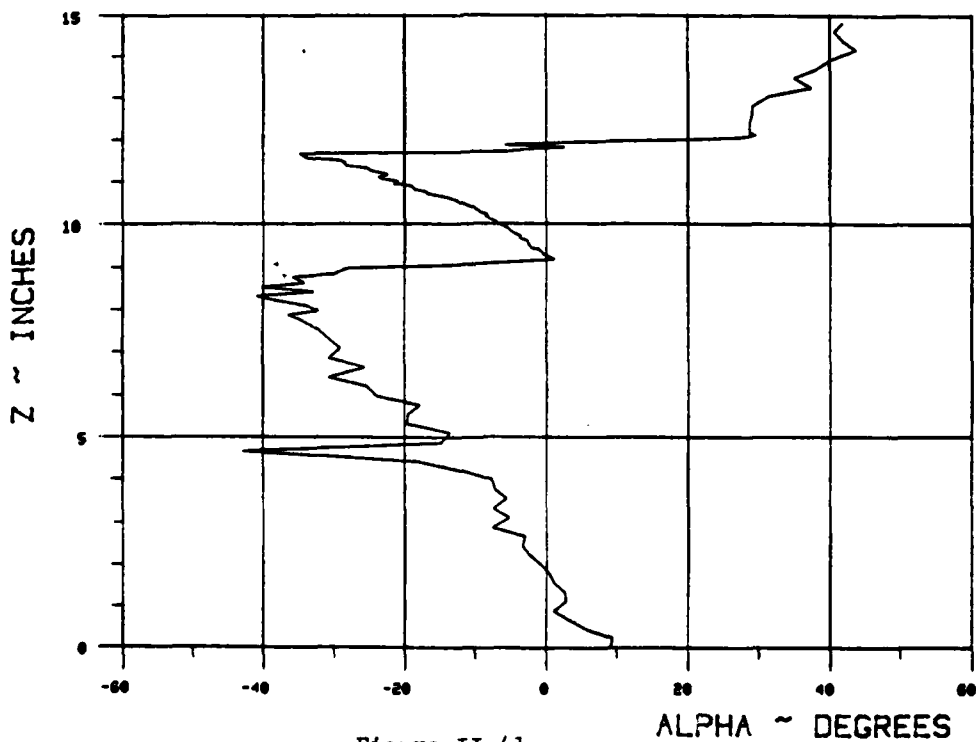
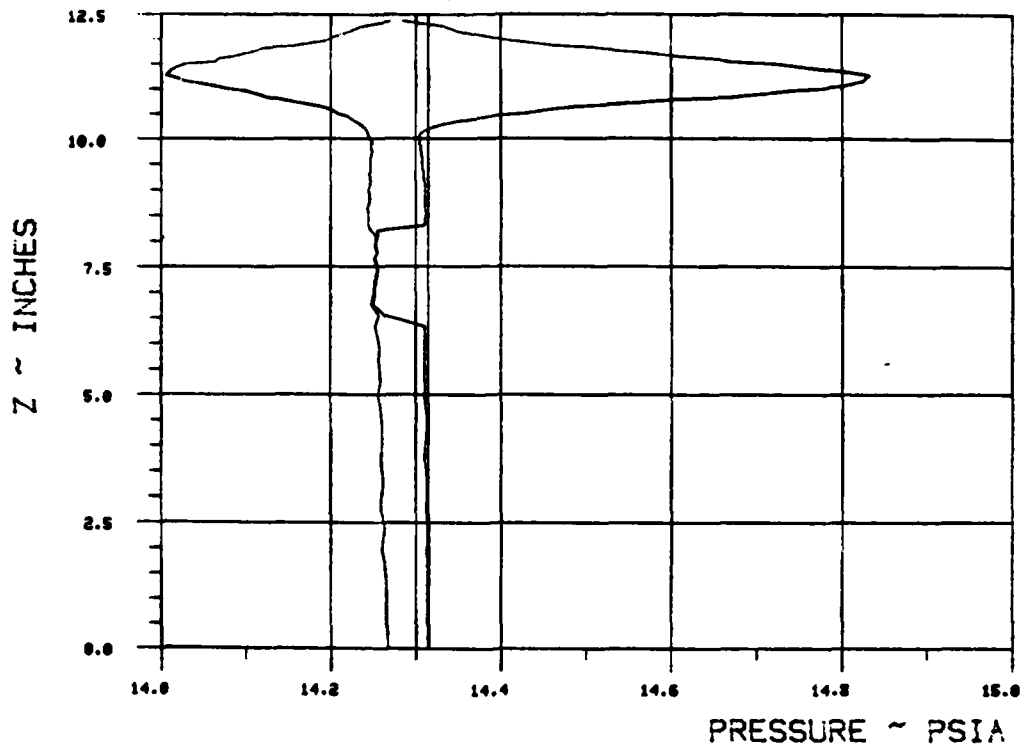


Figure II-41.

WAKE SURVEY DATA

TEST 272 RUN 49

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 11.4979



$X_{ss}/\text{chord} = .1914$ $Y_{ss}/\text{chord} = .0573$

BETA = 45.2

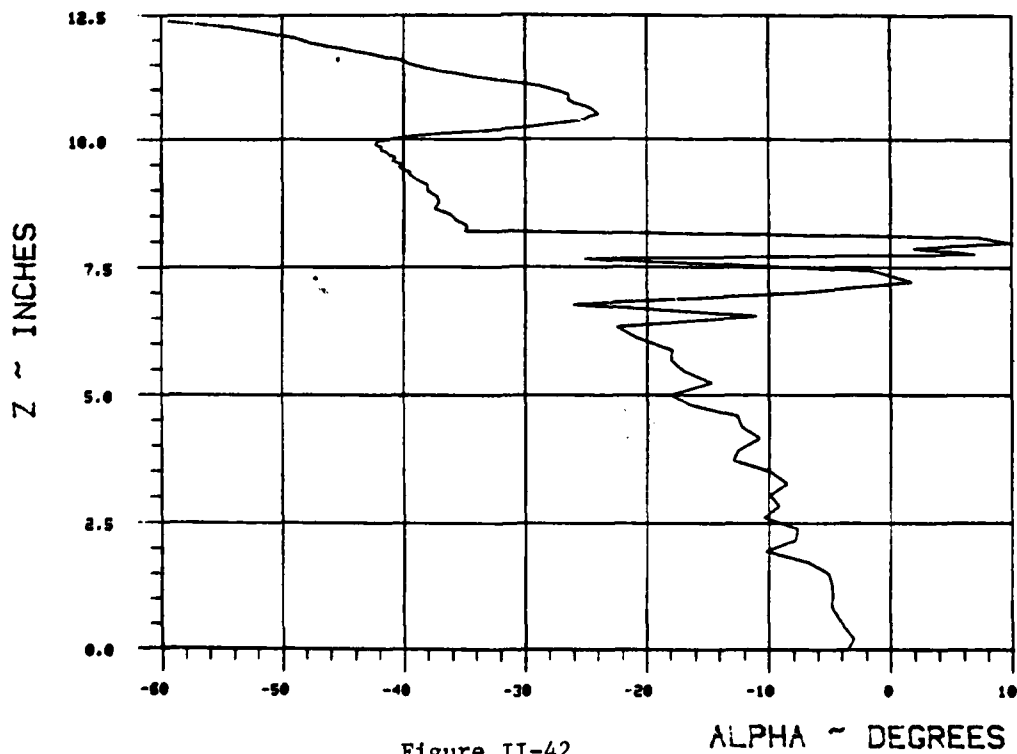
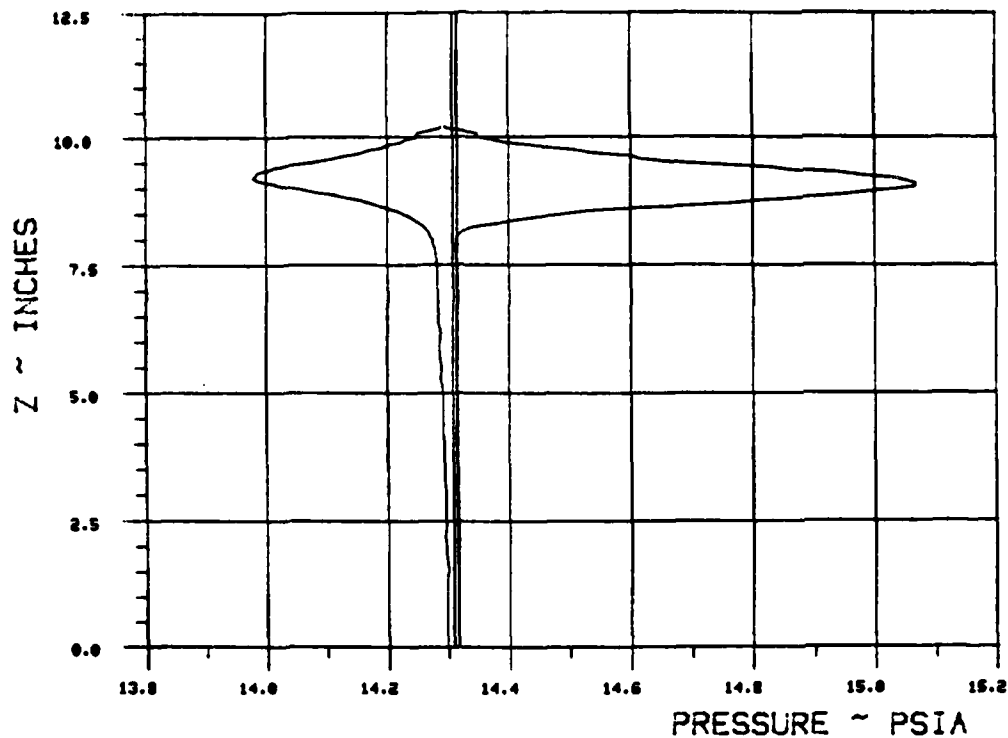


Figure II-42.

WAKE SURVEY DATA

TEST 272 RUN 54

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 11.803



$X_{ee}/\text{chord} = .1839$ $Y_{ee}/\text{chord} = .0335$

BETA = 55

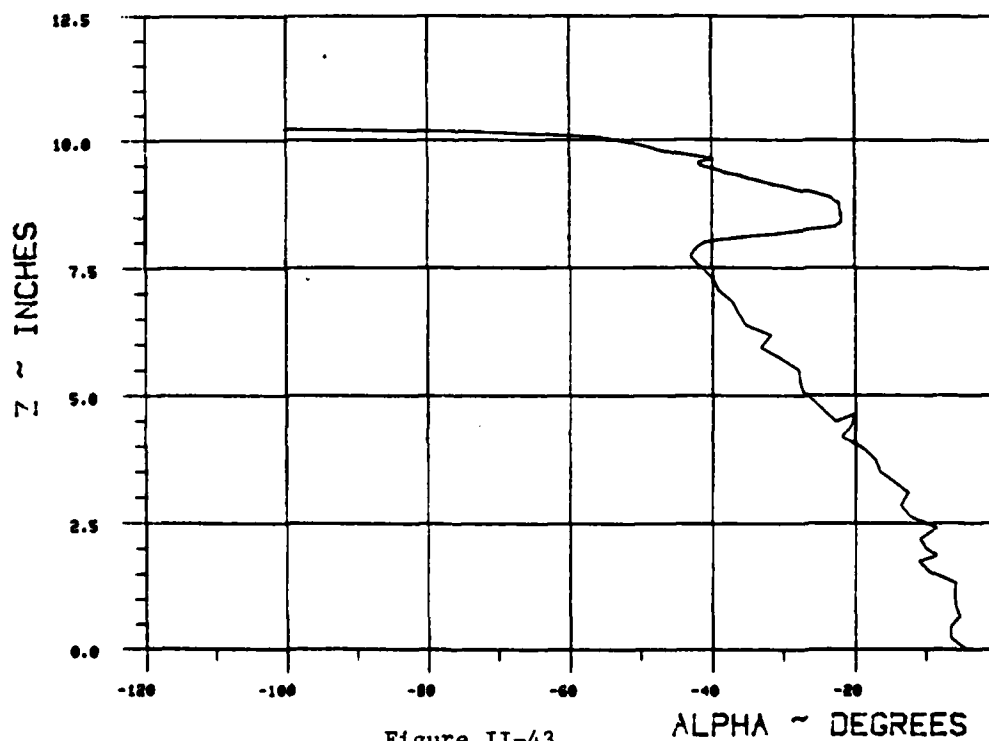
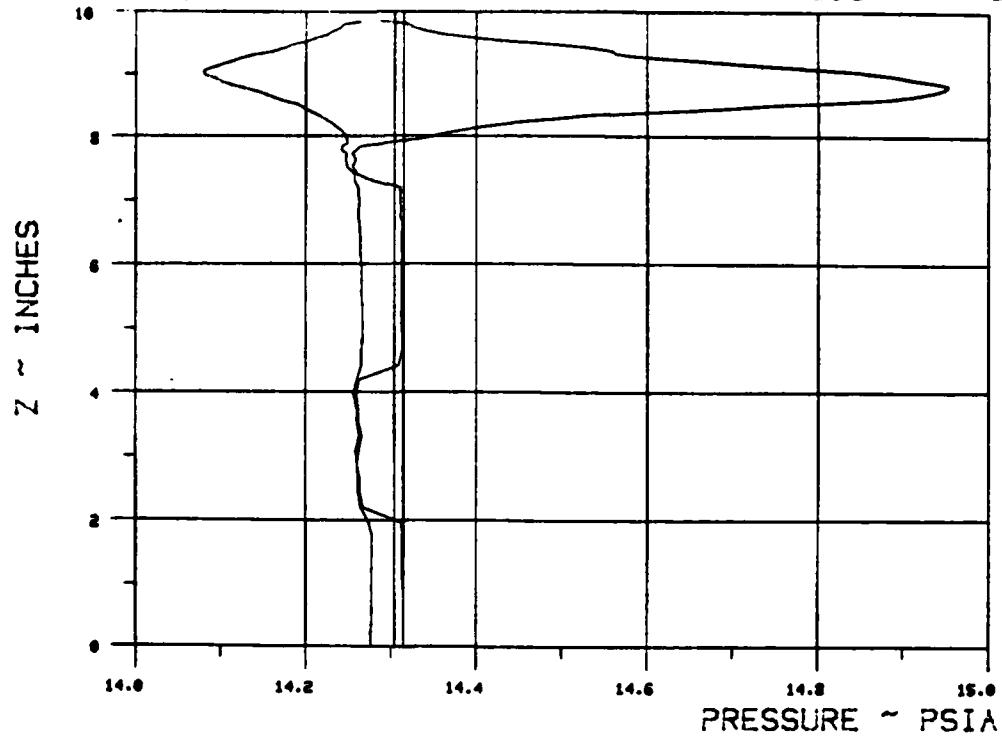


Figure II-43.

WAKE SURVEY DATA

TEST 272 RUN 51

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 15.2237



$X_{ss}/\text{chord} = .1385$ $Y_{ss}/\text{chord} = .0017$

BETA = 55

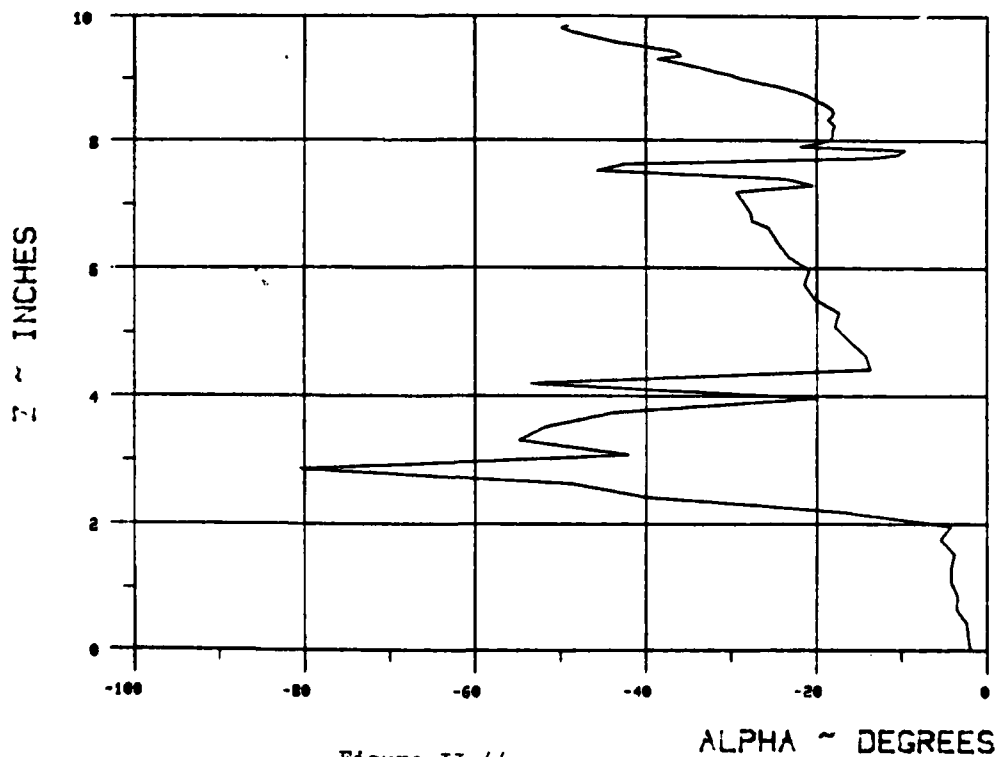
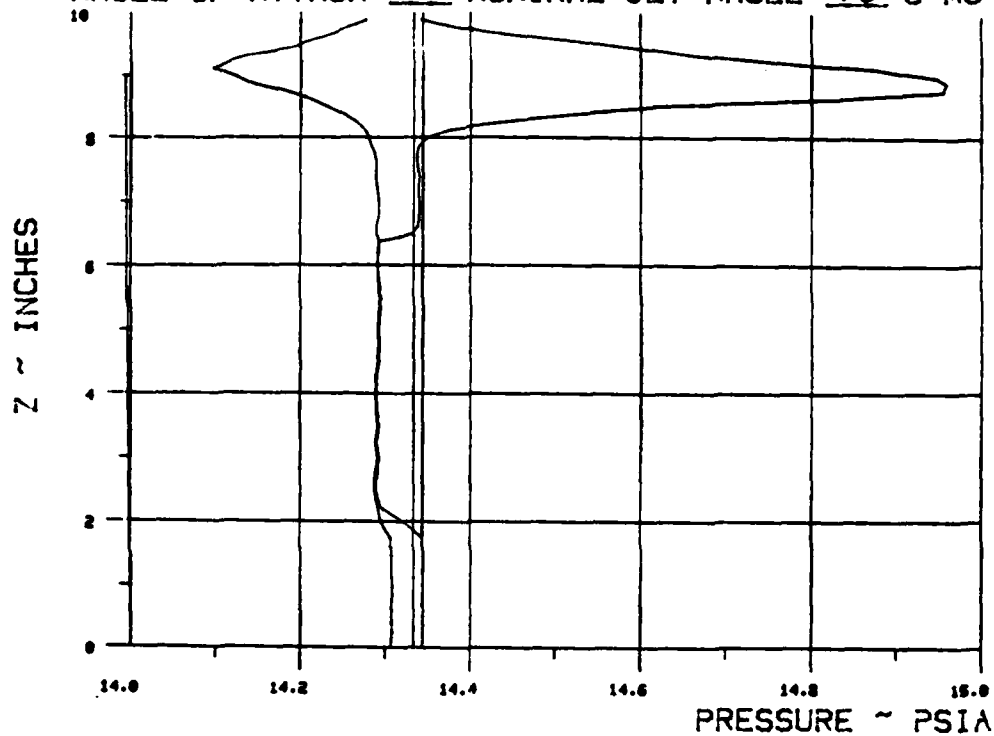


Figure II-44.

WAKE SURVEY DATA

TEST 272 RUN 55

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 15.3477



$X_{ss}/\text{chord} = .138$ $Y_{ss}/\text{chord} = .0014$

BETA = 55

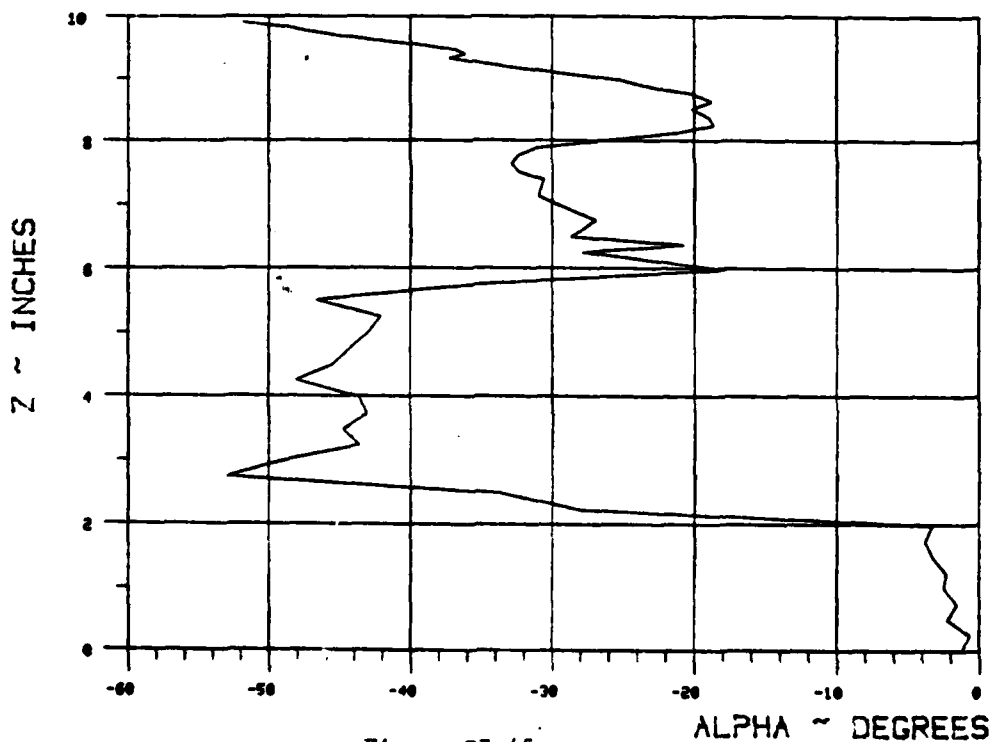
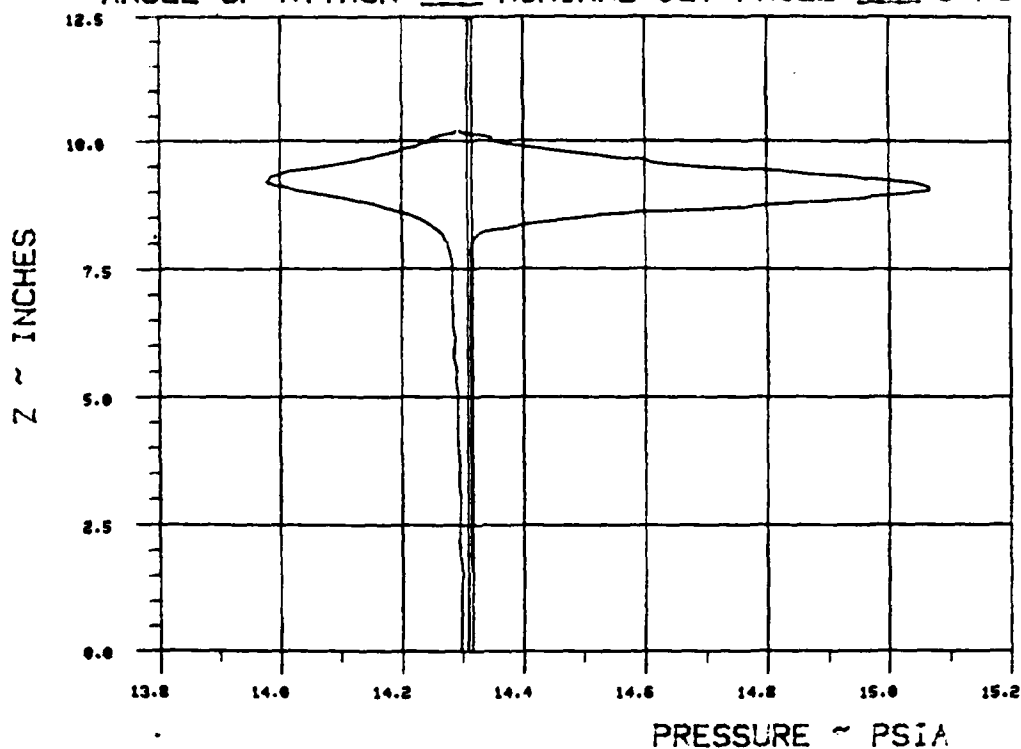


Figure II-45.

WAKE SURVEY DATA

TEST 272 RUN 52

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 25.5336



$X_{ss}/\text{chord} = .1384$ $Y_{ss}/\text{chord} = .0016$

BETA = 55

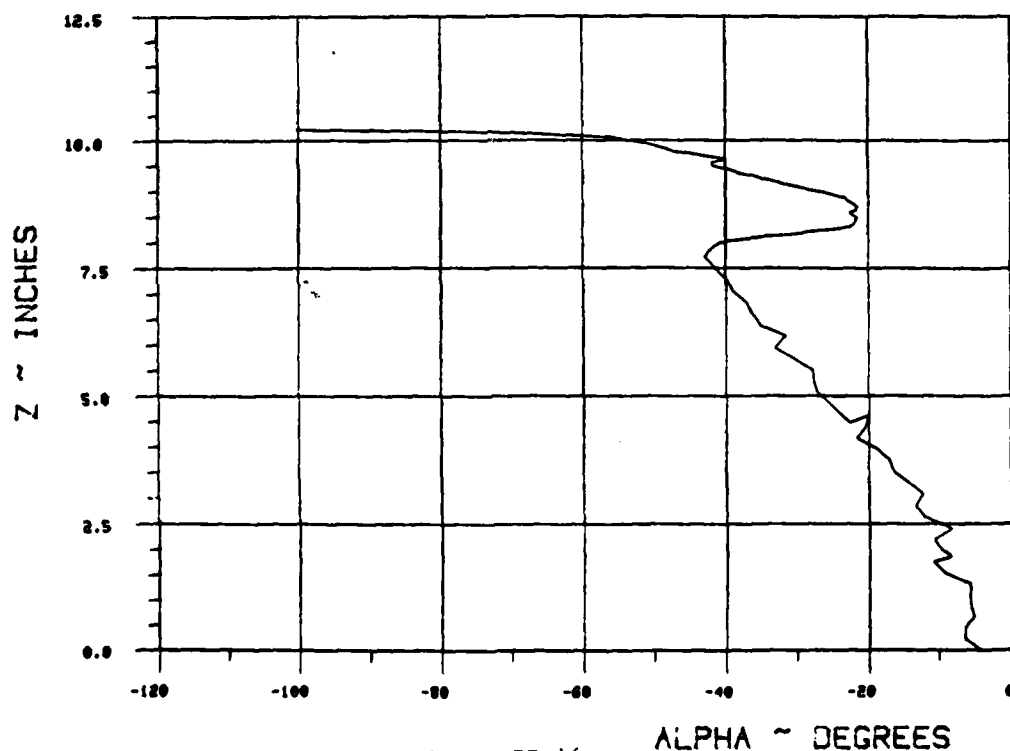


Figure II-46.

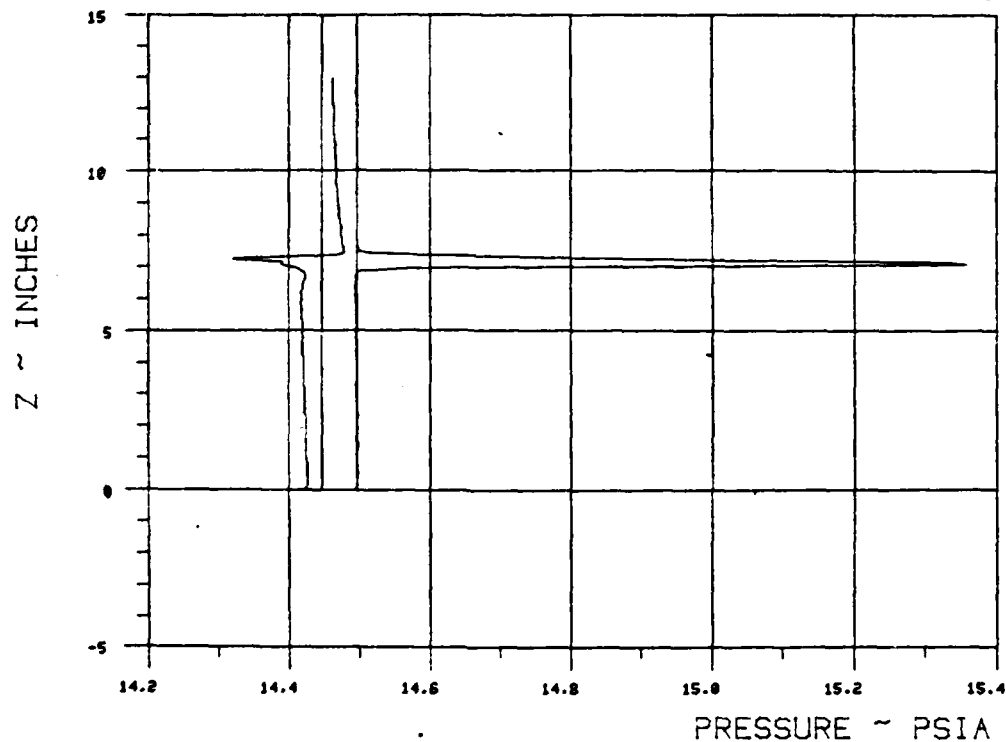
APPENDIX III

WAKE SURVEY DATA

WAKE SURVEY DATA

TEST 287 RUN 42

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 1.0409



$X_{ss}/chord = .29$ $Y_{ss}/chord = .76$

BETA = 10

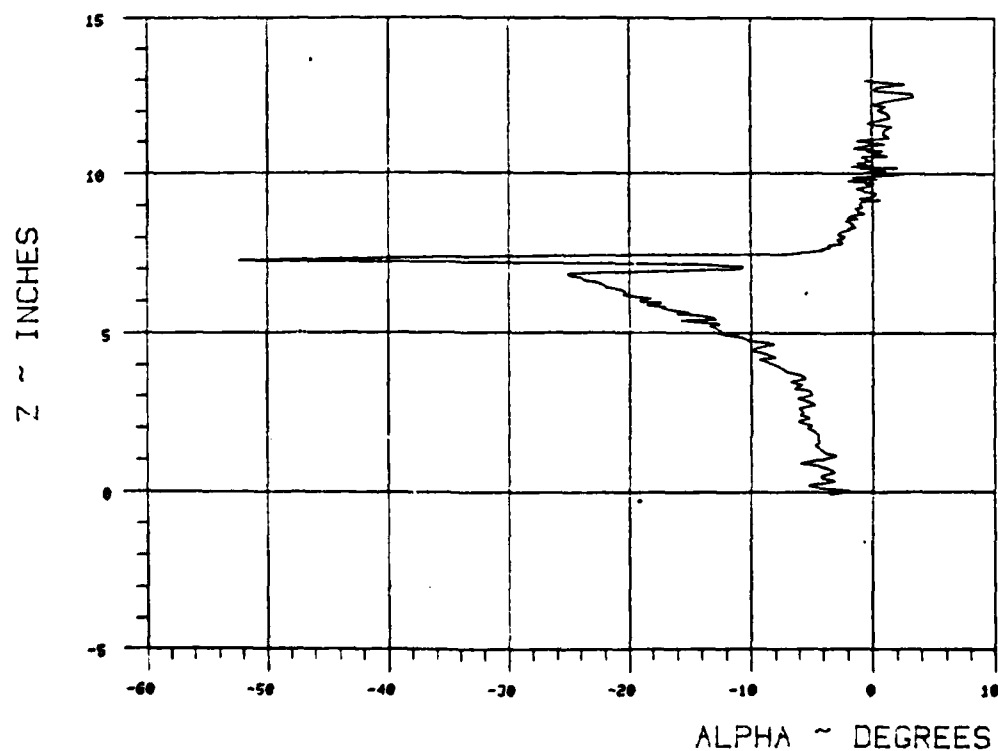
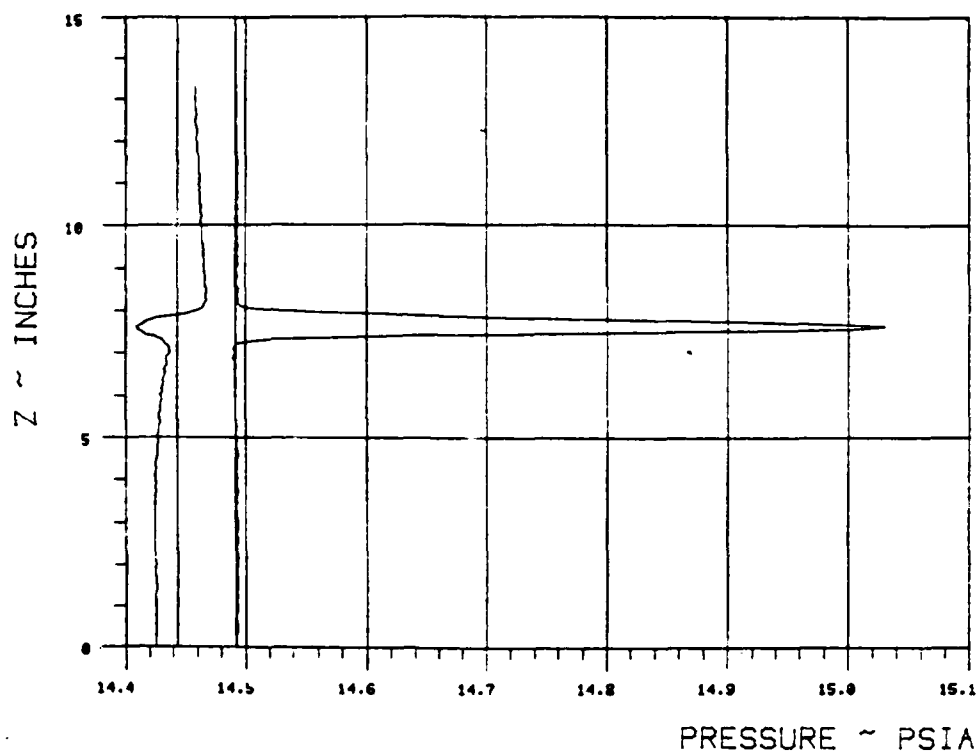


Figure III-1.

WAKE SURVEY DATA

TEST 287 RUN 43

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 1.0286



$X_{ss}/\text{chord} = \underline{.413}$ $Y_{ss}/\text{chord} = \underline{.738}$

BETA = 10

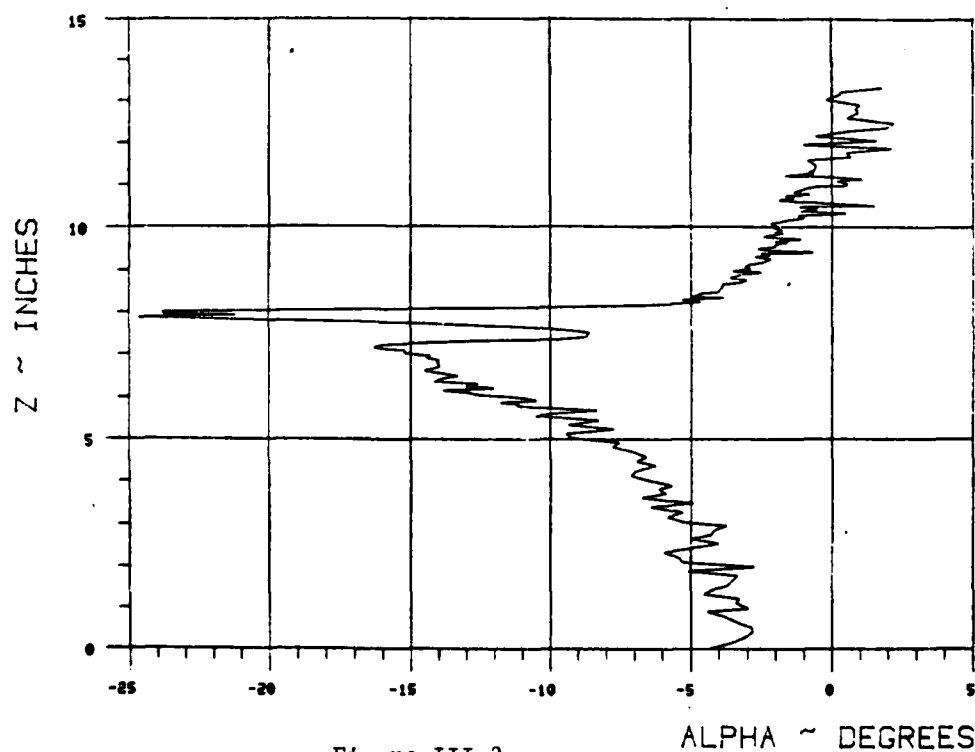


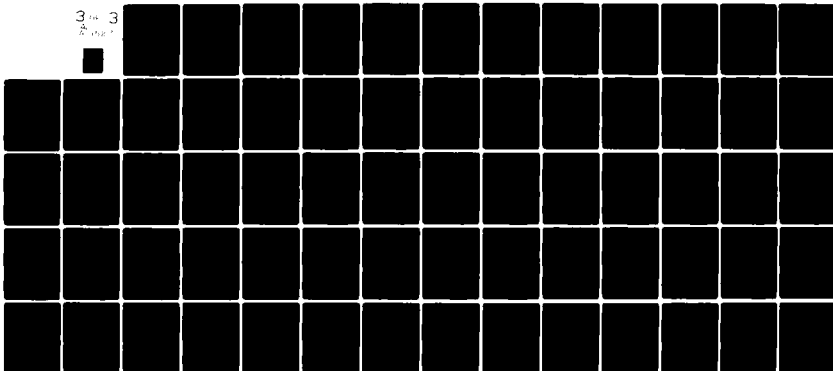
Figure III-2.

AD-A111 527

ROCKWELL INTERNATIONAL COLUMBUS OH NORTH AMERICAN AI-ETC F/G 20/4
PROGRESS TOWARDS A THEORY OF JET FLAP THRUST RECOVERY. (U)
SEP 80 P M DEVILAGUA, P E COLE, E F SCHUM F49620-78-C-0069
NR88H-76 AFOSR-TR-81-0900 NL

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3 of 3
Page 1



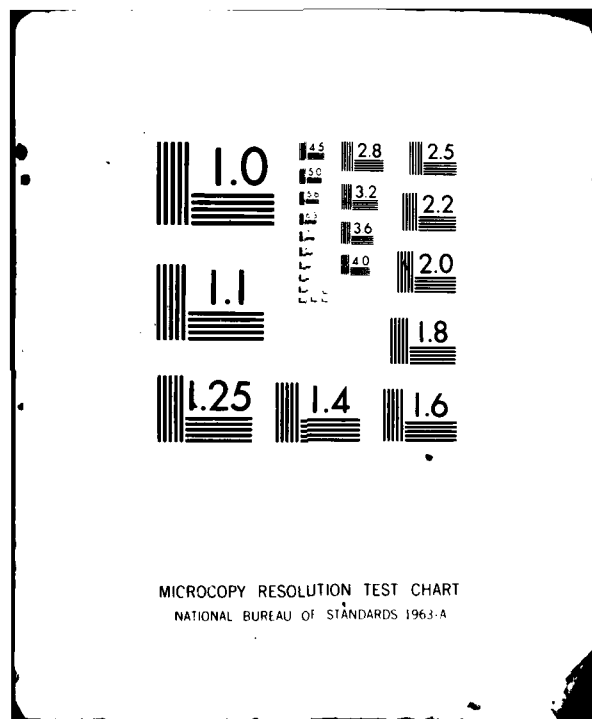
END

DATE

FILED

82

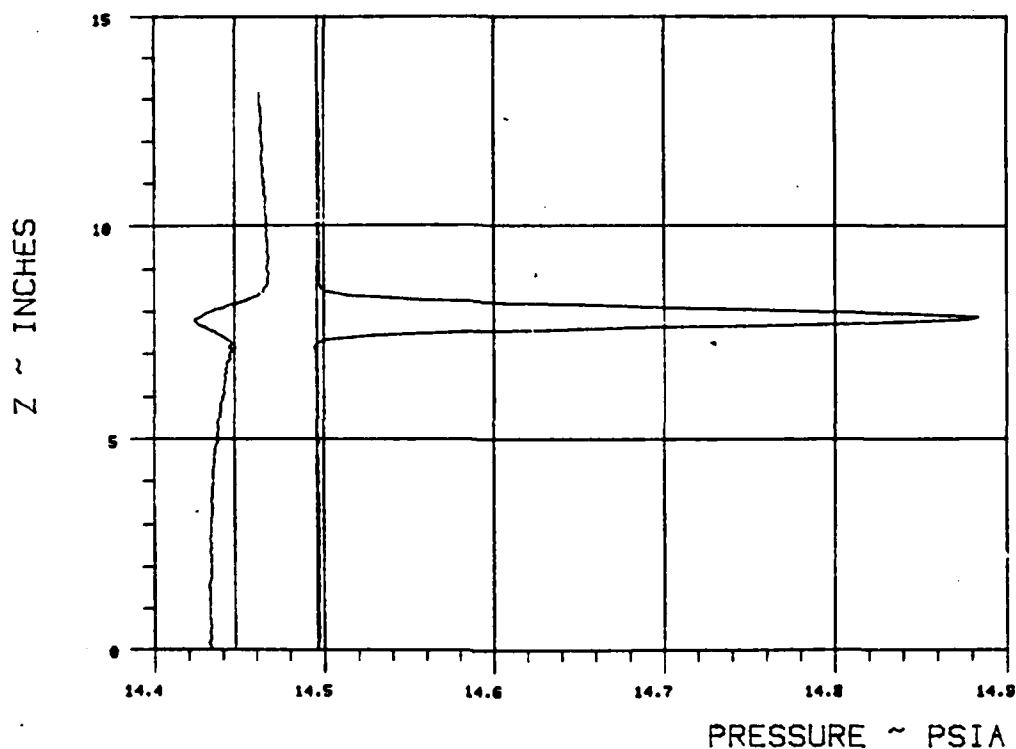
DTIC



WAKE SURVEY DATA

TEST 287 RUN 44

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 1.0395



$X_{ss}/\text{chord} = \underline{.536}$ $Y_{ss}/\text{chord} = \underline{.717}$

BETA = 10

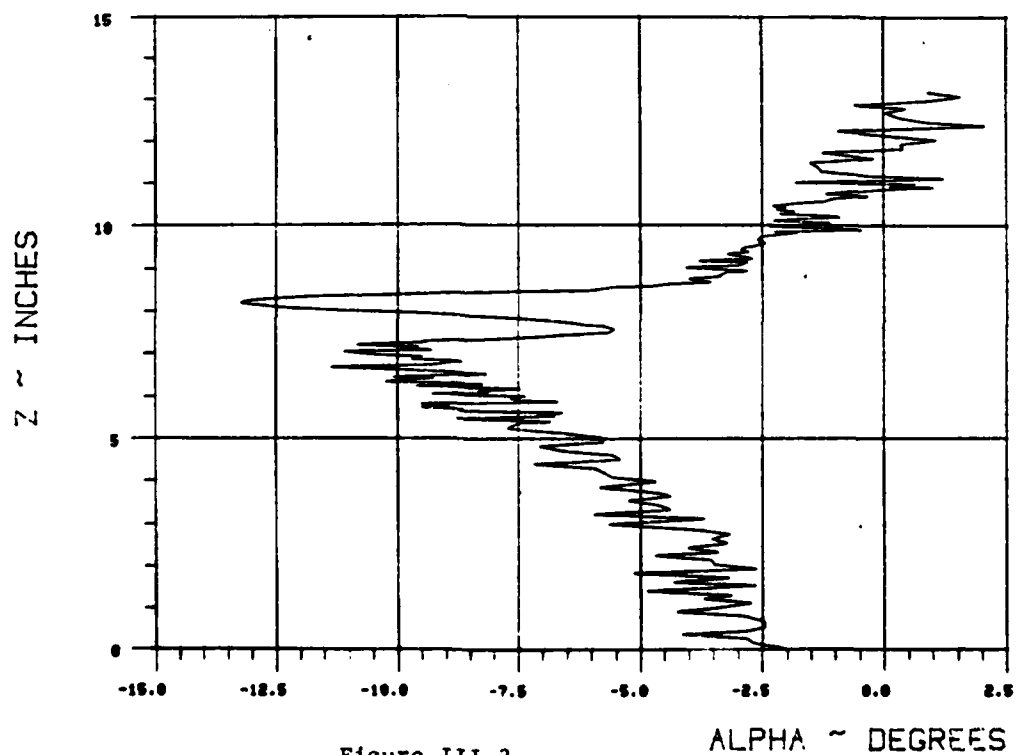
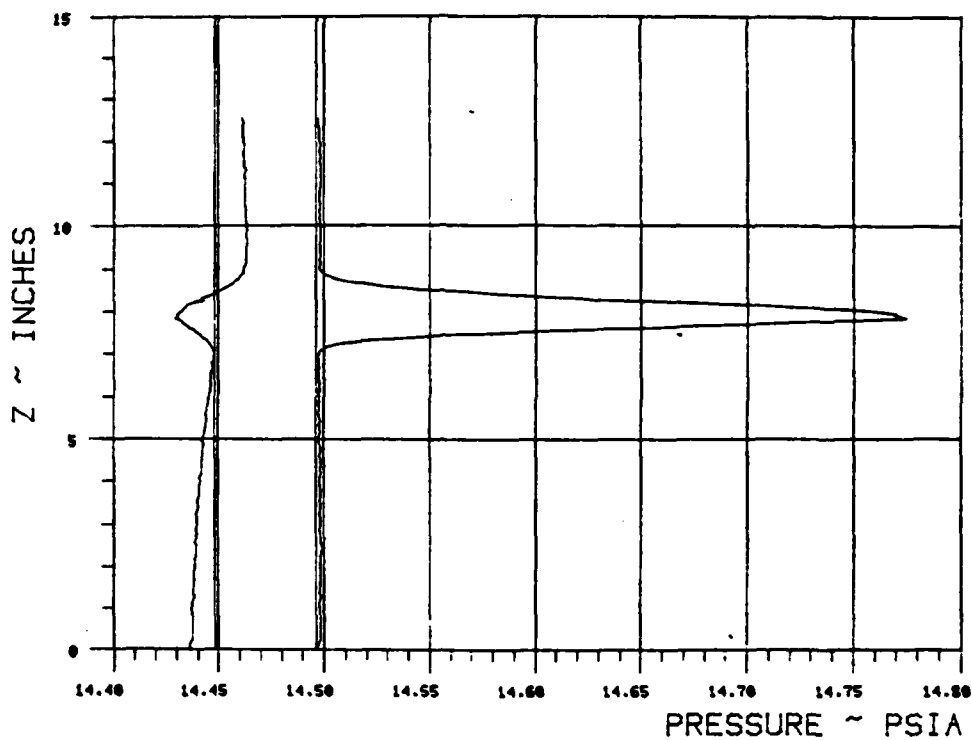


Figure III-3.

WAKE SURVEY DATA

TEST 287 RUN 45

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 1.059



$X_{ss}/\text{chord} = .783$ $Y_{ss}/\text{chord} = .673$

BETA = 10

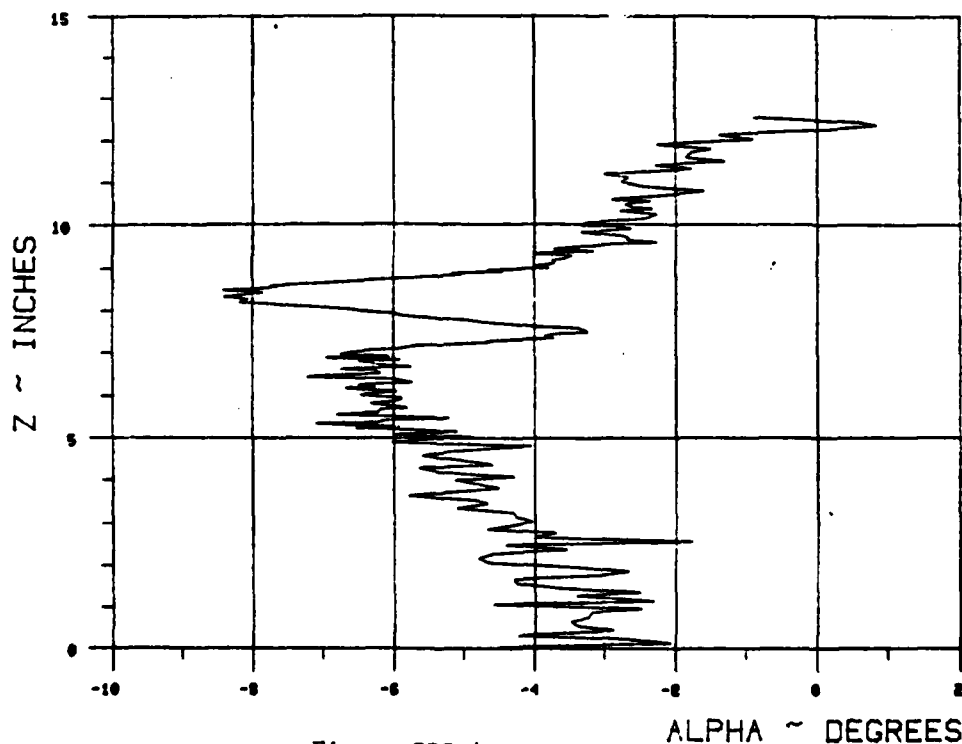
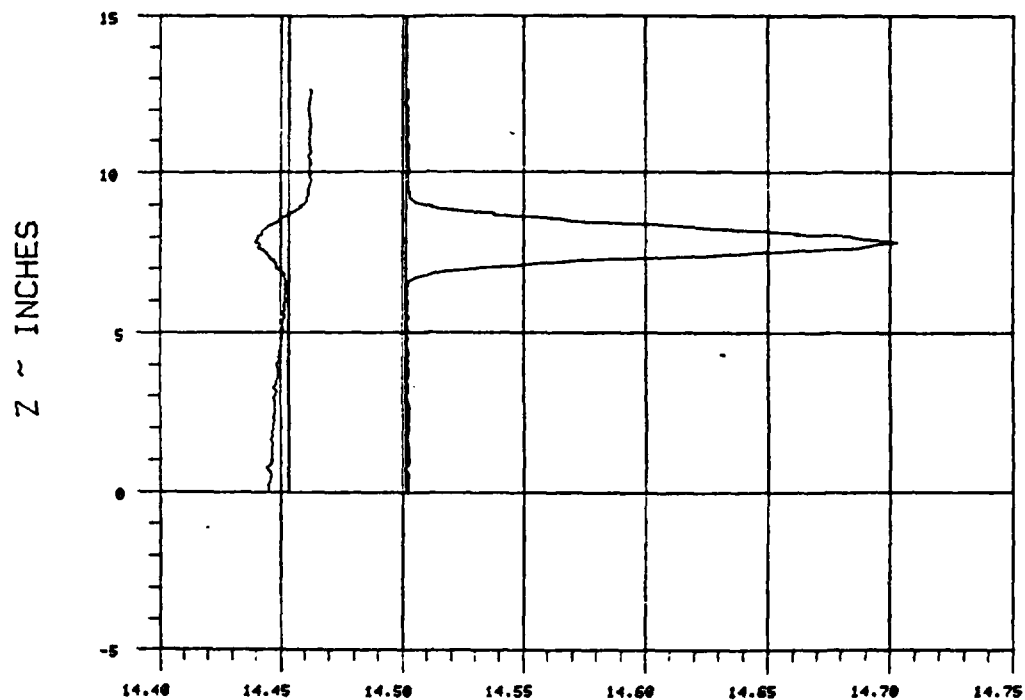


Figure III-4.

WAKE SURVEY DATA

TEST 287 RUN 46

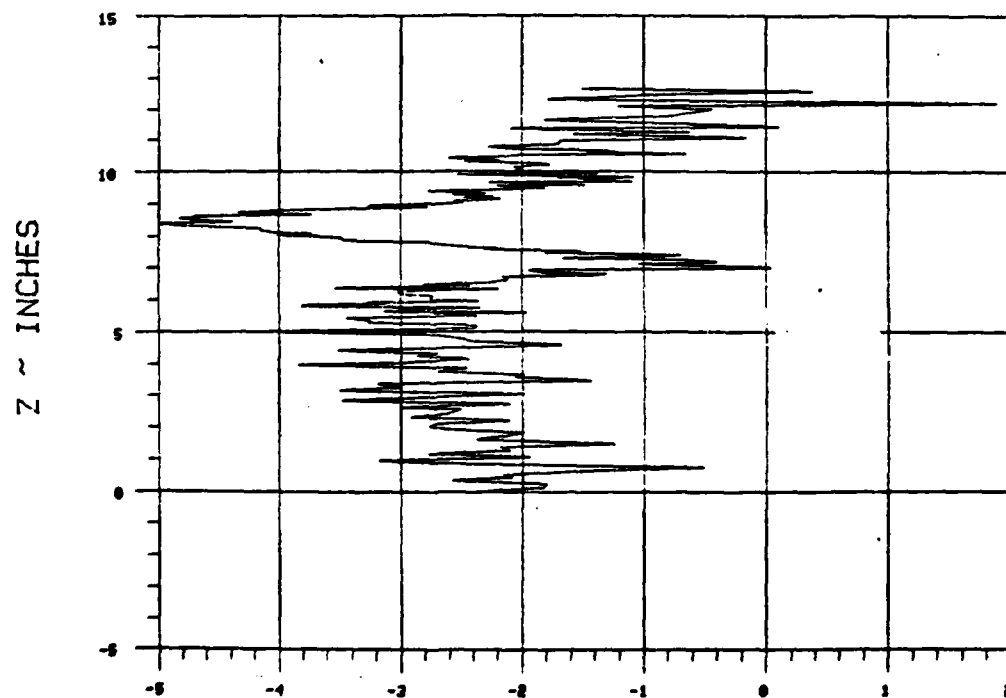
ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 1.0268



PRESSURE ~ PSIA.

$X_{ss}/\text{chord} = \underline{1.152}$ $Y_{ss}/\text{chord} = \underline{.608}$

BETA = 10



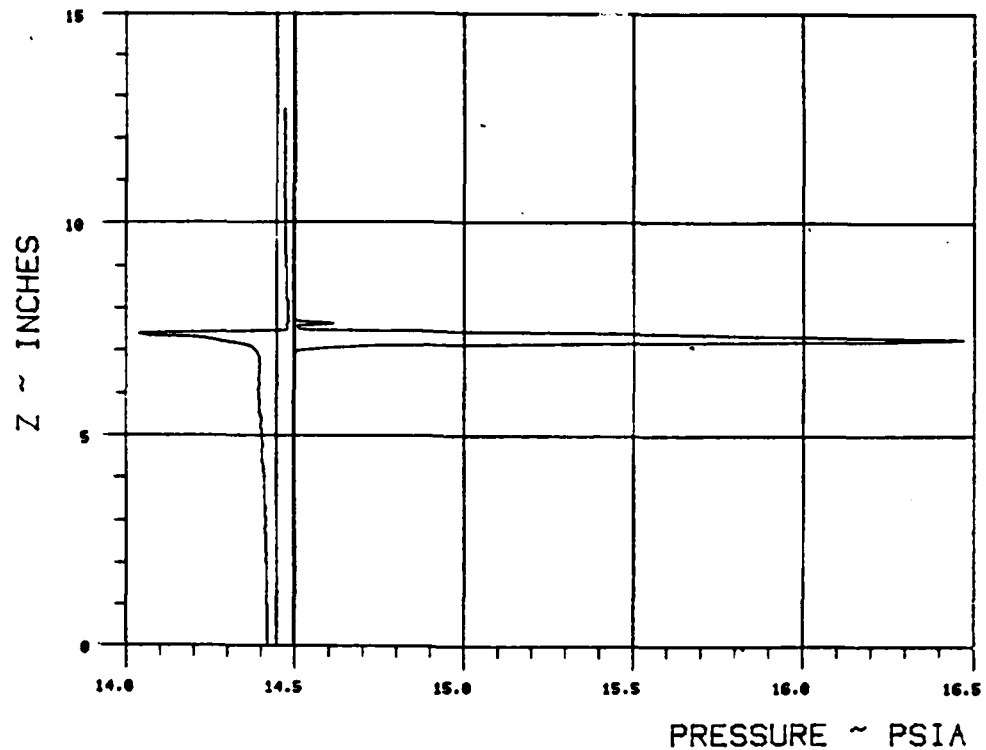
ALPHA ~ DEGREES

Figure III-5.

WAKE SURVEY DATA

TEST 287 RUN 47

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 2.098



$X_{ss}/\text{chord} = \underline{.29}$ $Y_{ss}/\text{chord} = \underline{.76}$

BETA = 10

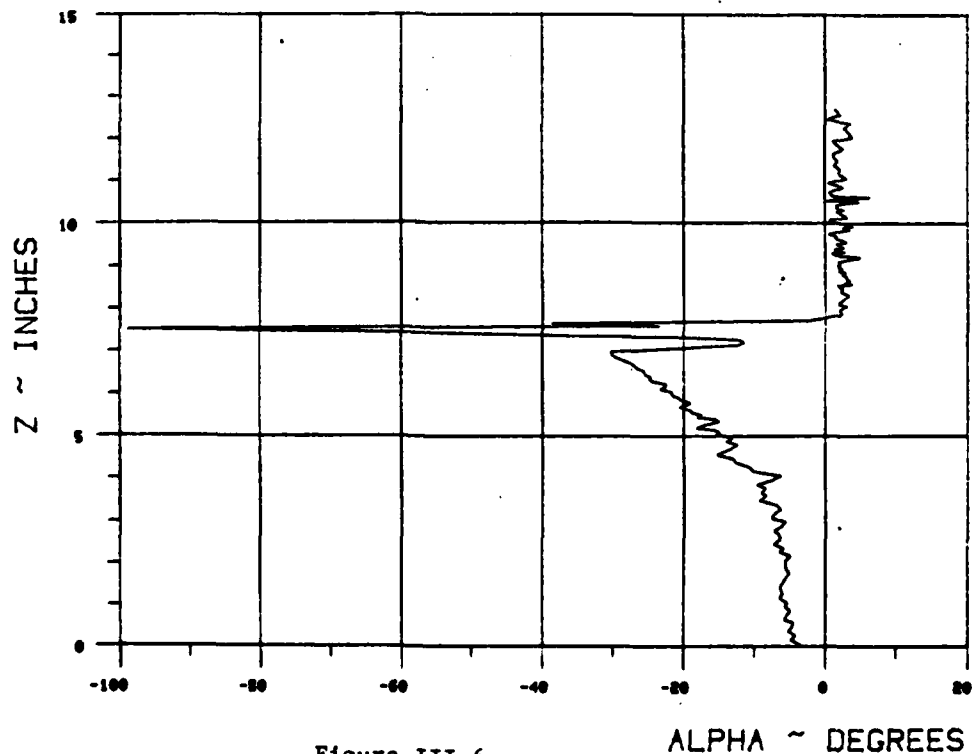
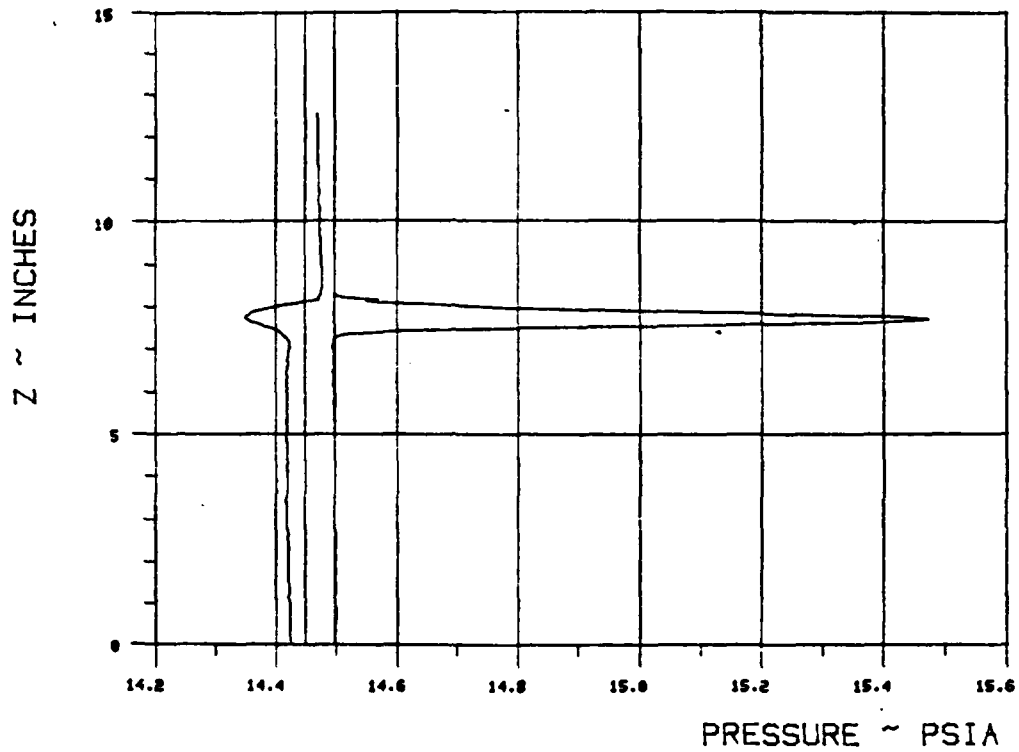


Figure III-6.

WAKE SURVEY DATA

TEST 287 RUN 48

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 2.0862



$X_{se}/\text{chord} = \underline{.413}$ $Y_{se}/\text{chord} = \underline{.738}$

BETA = 10

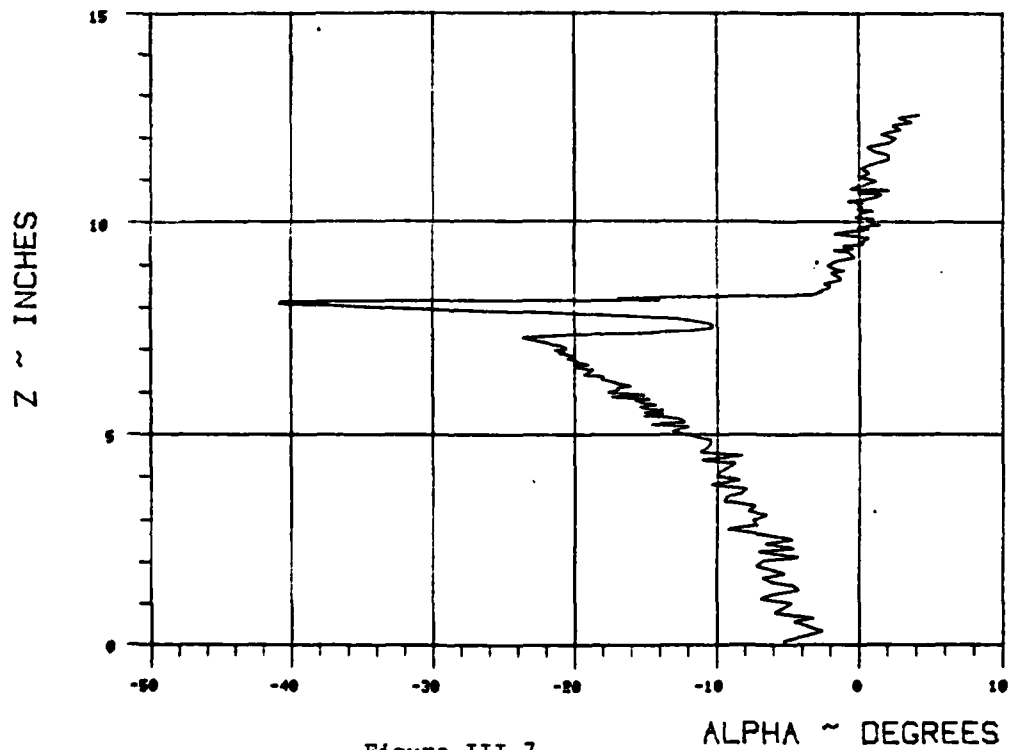
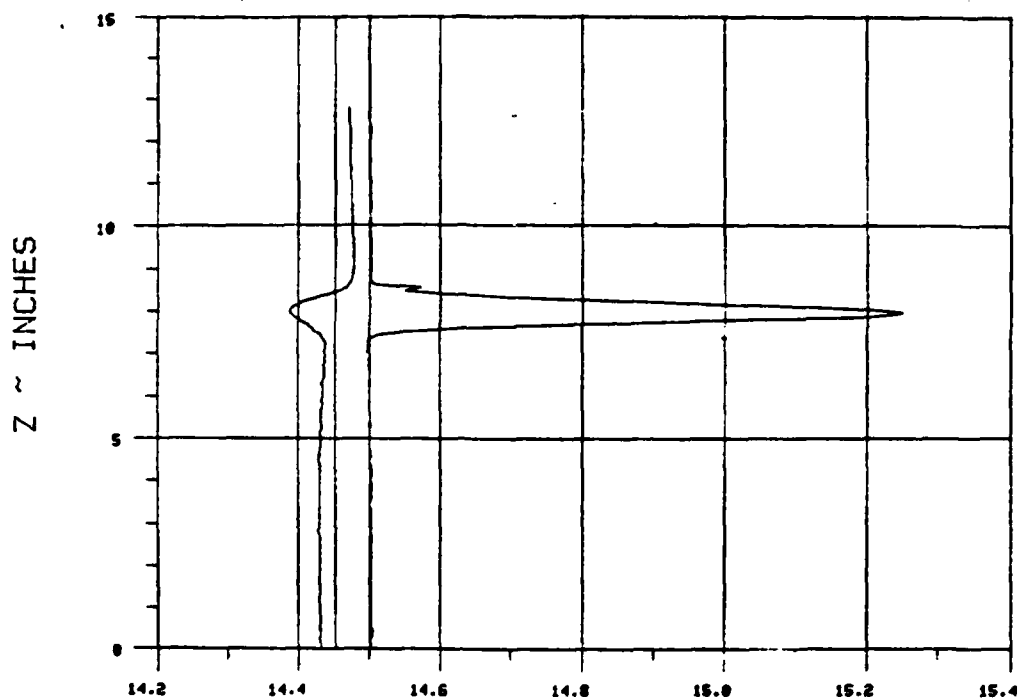


Figure III-7.

WAKE SURVEY DATA

TEST 287 RUN 49

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 2.0959



PRESSURE ~ PSIA

$X_{ss}/chord = .536$ $Y_{ss}/chord = .717$

BETA = 10

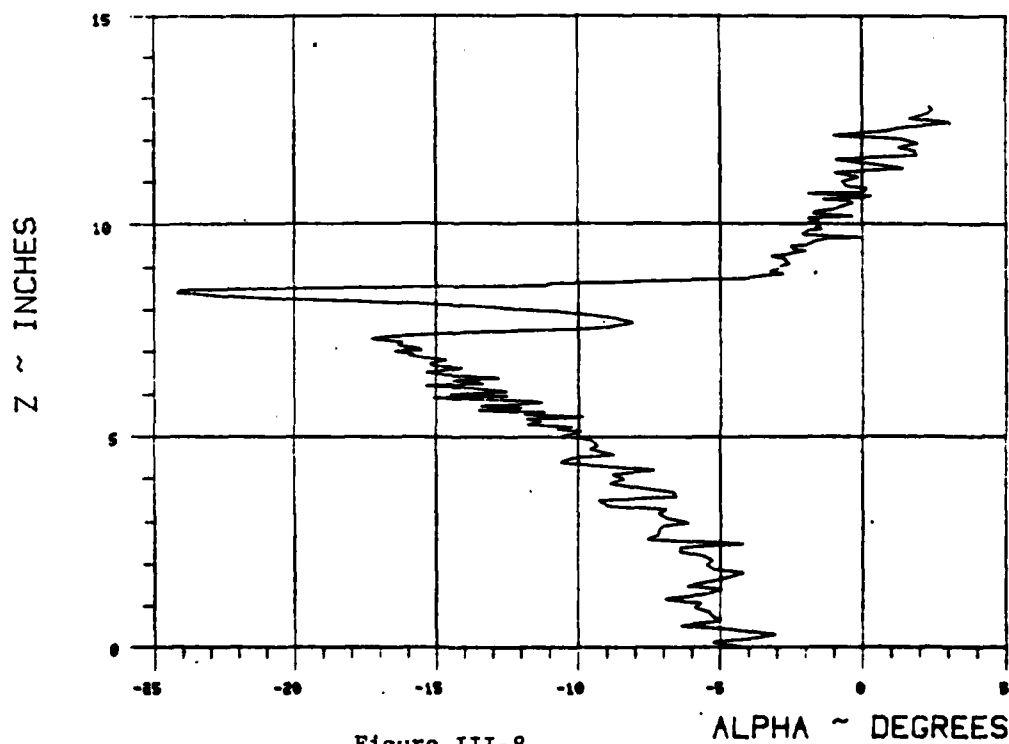


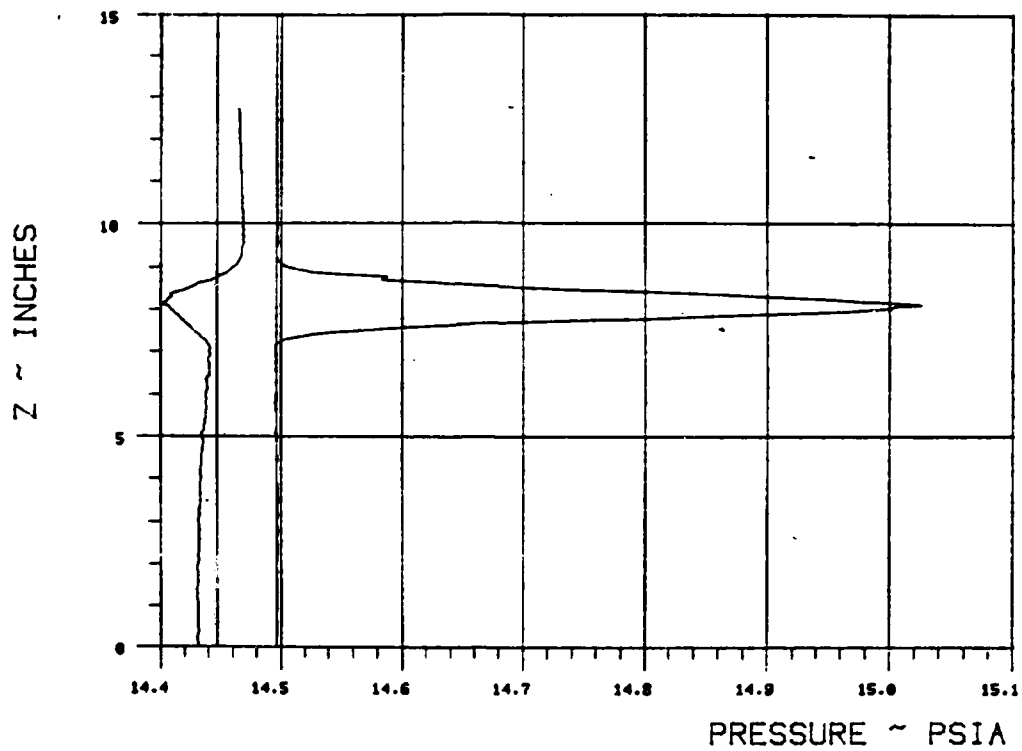
Figure III-8.

ALPHA ~ DEGREES

WAKE SURVEY DATA

TEST 287 RUN 50

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 2.0595



$X_{ss}/\text{chord} = \underline{.783}$ $Y_{ss}/\text{chord} = \underline{.673}$

BETA = 10

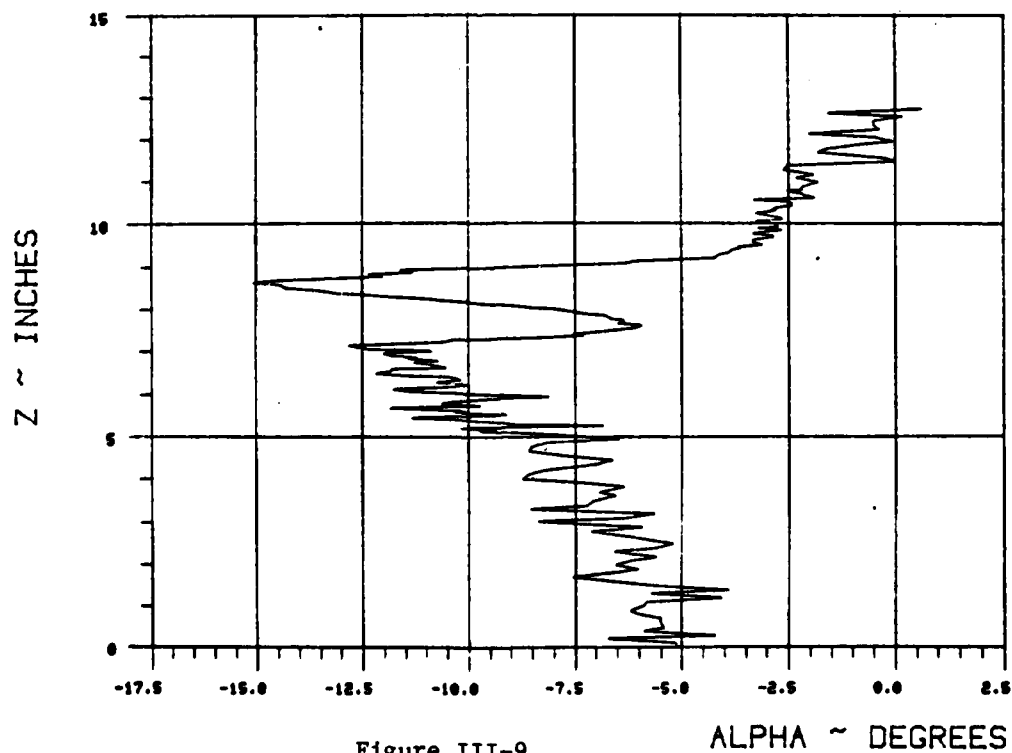
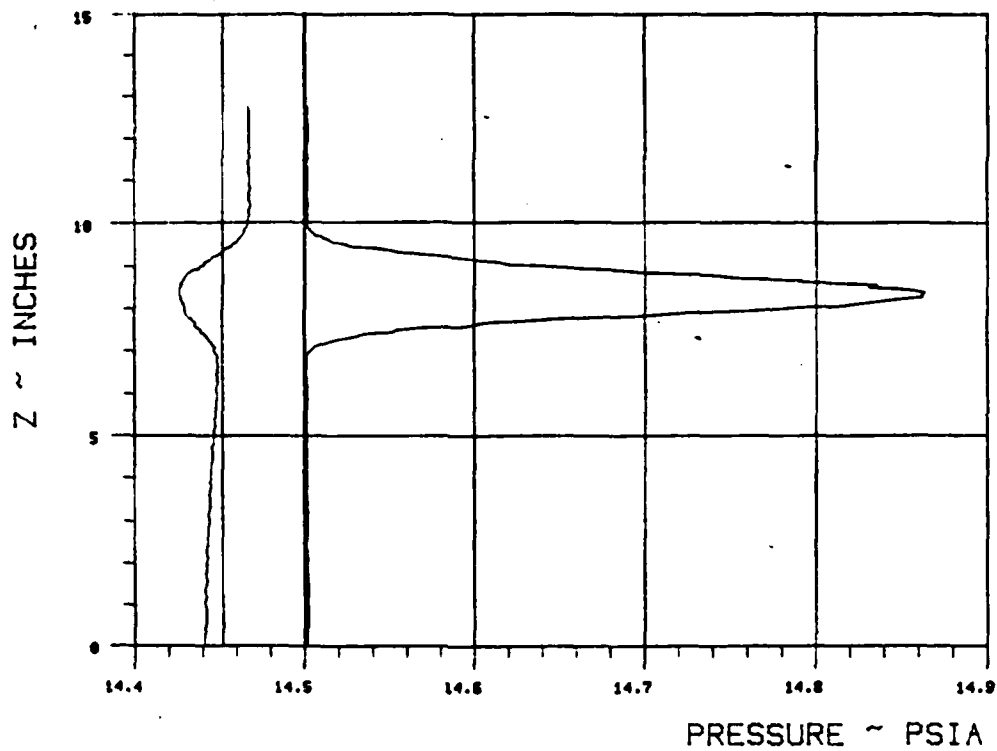


Figure III-9.

WAKE SURVEY DATA

TEST 287 RUN 51

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 2.0646



$X_{ss}/\text{chord} = \underline{1.152}$ $Y_{ss}/\text{chord} = \underline{.608}$

BETA = 10

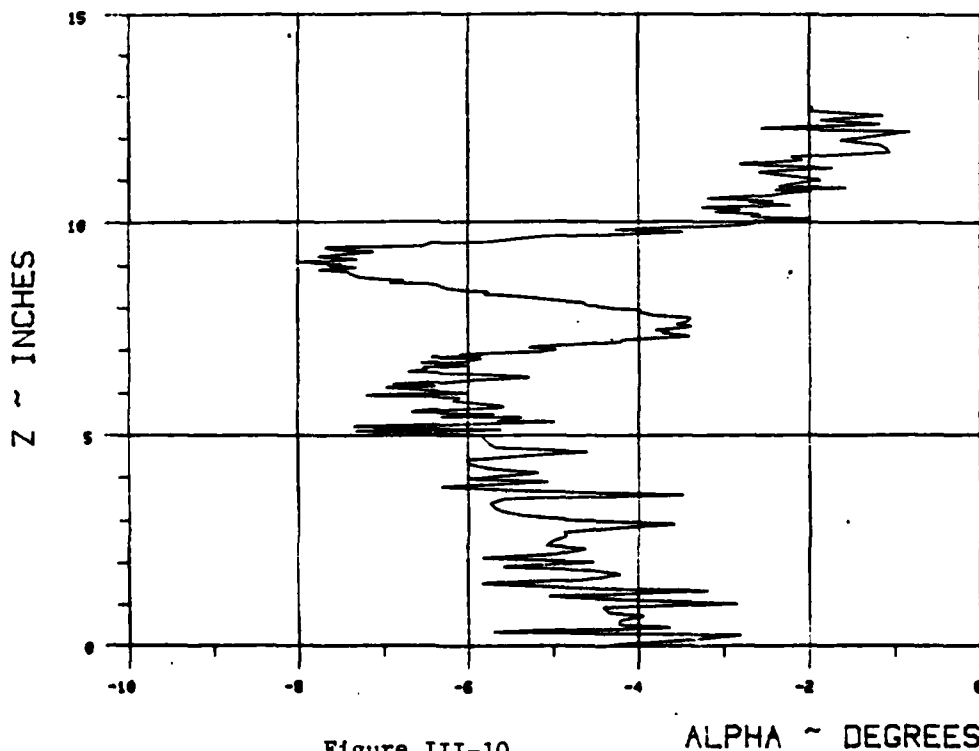
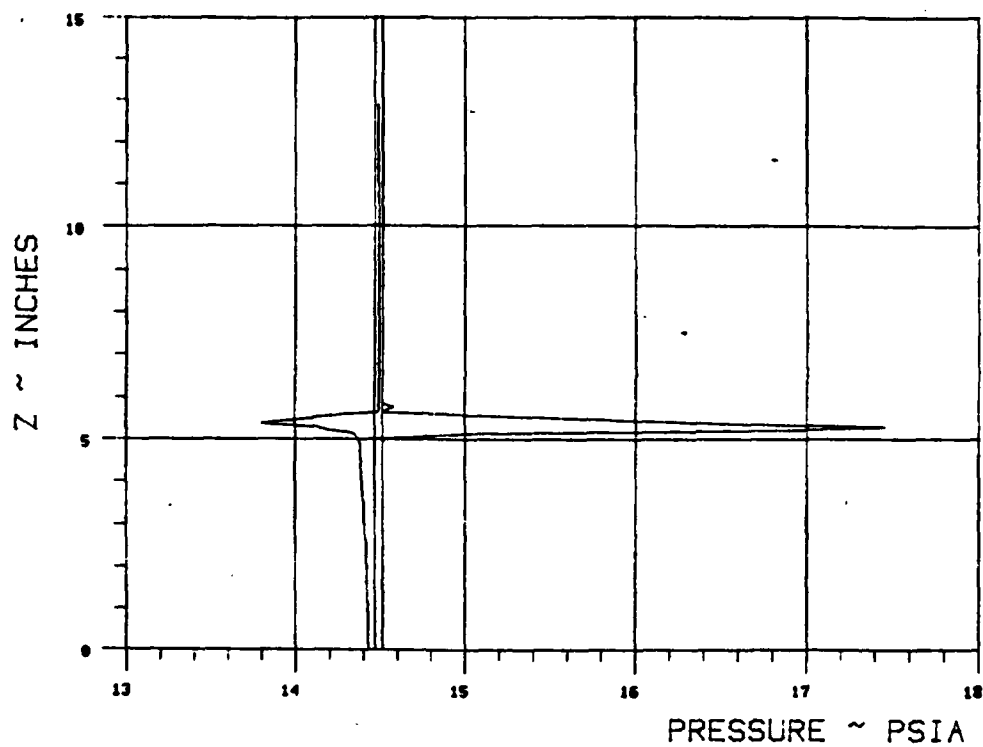


Figure III-10.

WAKE SURVEY DATA

TEST 287 RUN 52

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 4.1134



$X_{ss}/\text{chord} = \underline{.247}$ $Y_{ss}/\text{chord} = \underline{.514}$

BETA = 10

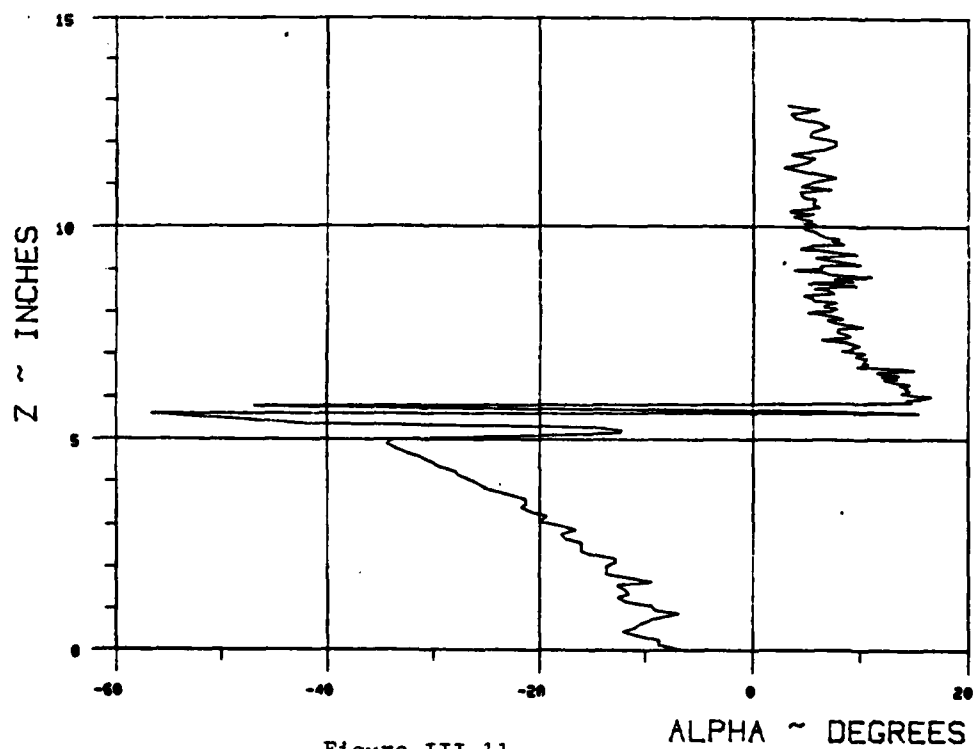
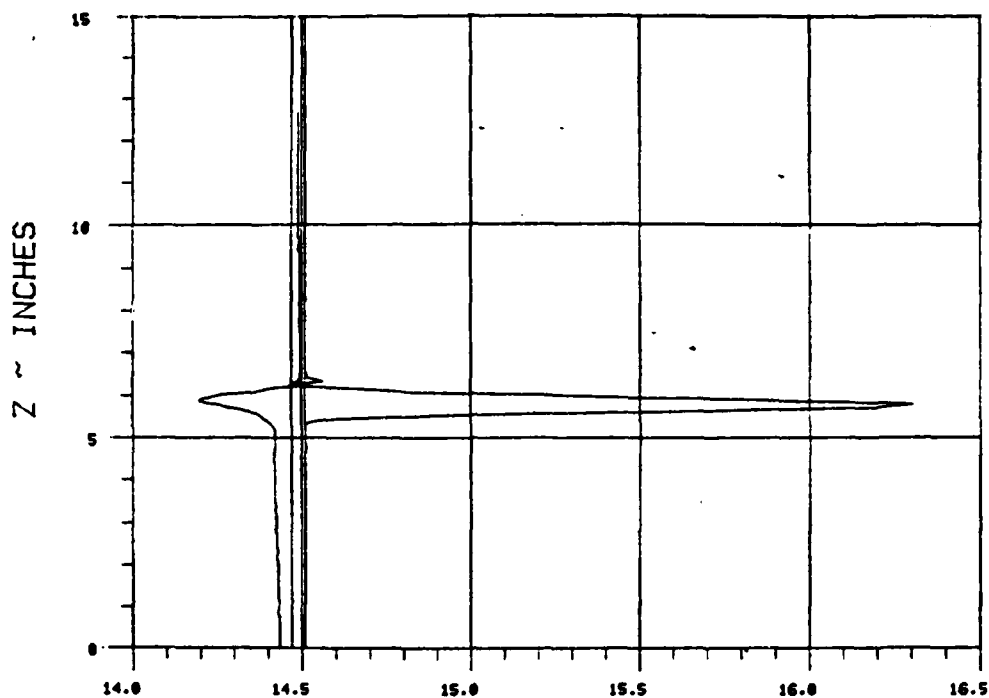


Figure III-11.

WAKE SURVEY DATA

TEST 287 RUN 53

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 4.1205



PRESSURE ~ PSIA

$X_{ss}/\text{chord} = .37$ $Y_{ss}/\text{chord} = .492$

BETA = 10

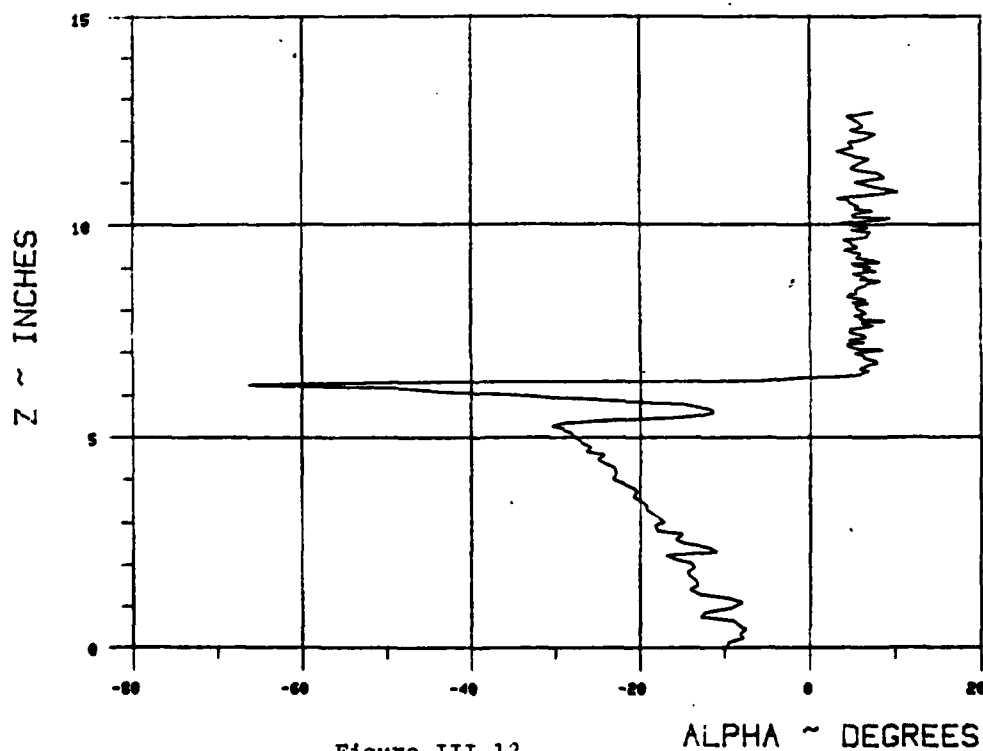


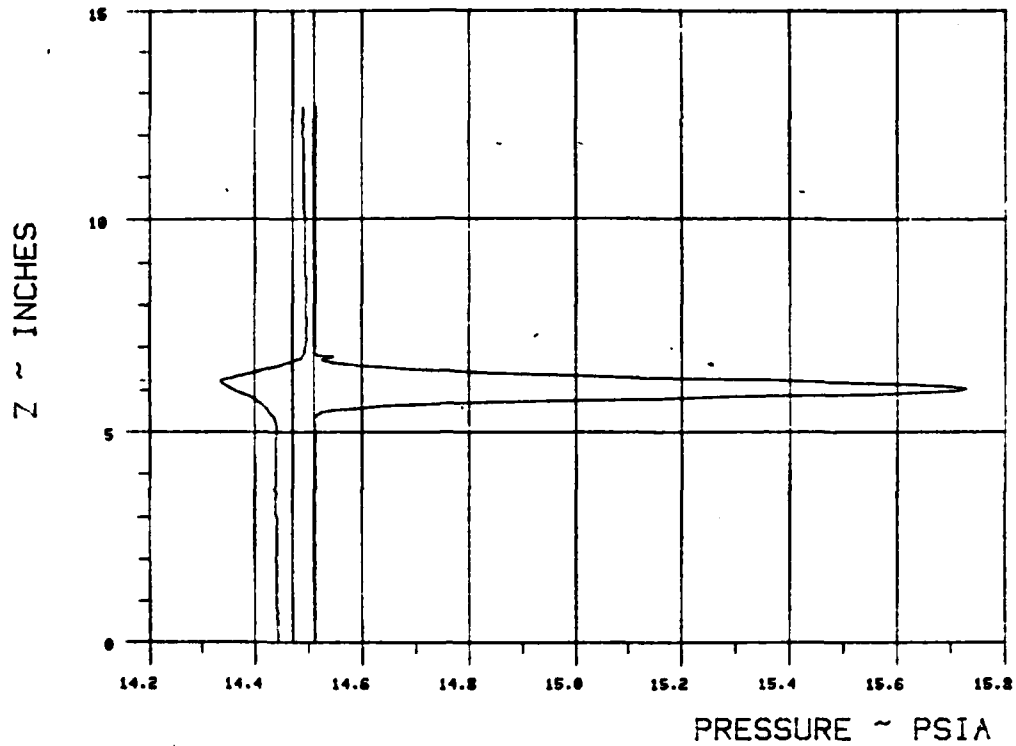
Figure III-12.

ALPHA ~ DEGREES

WAKE SURVEY DATA

TEST 287 RUN 54

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 4.1363



$X_{ss}/\text{chord} = .493$ $Y_{ss}/\text{chord} = .47$

BETA = 10

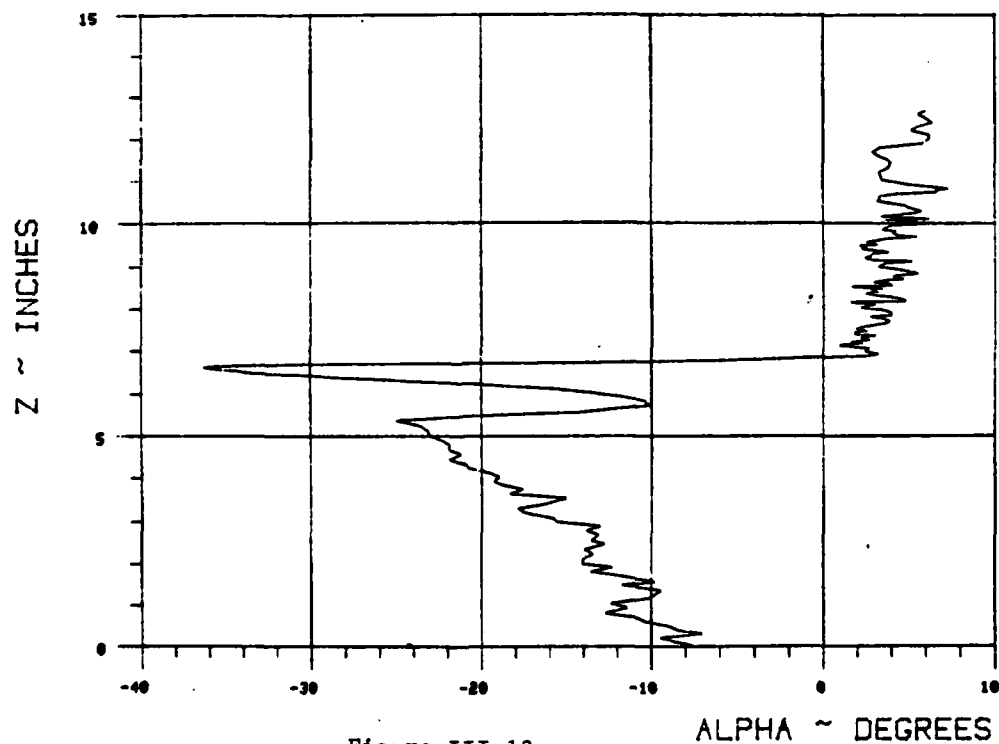
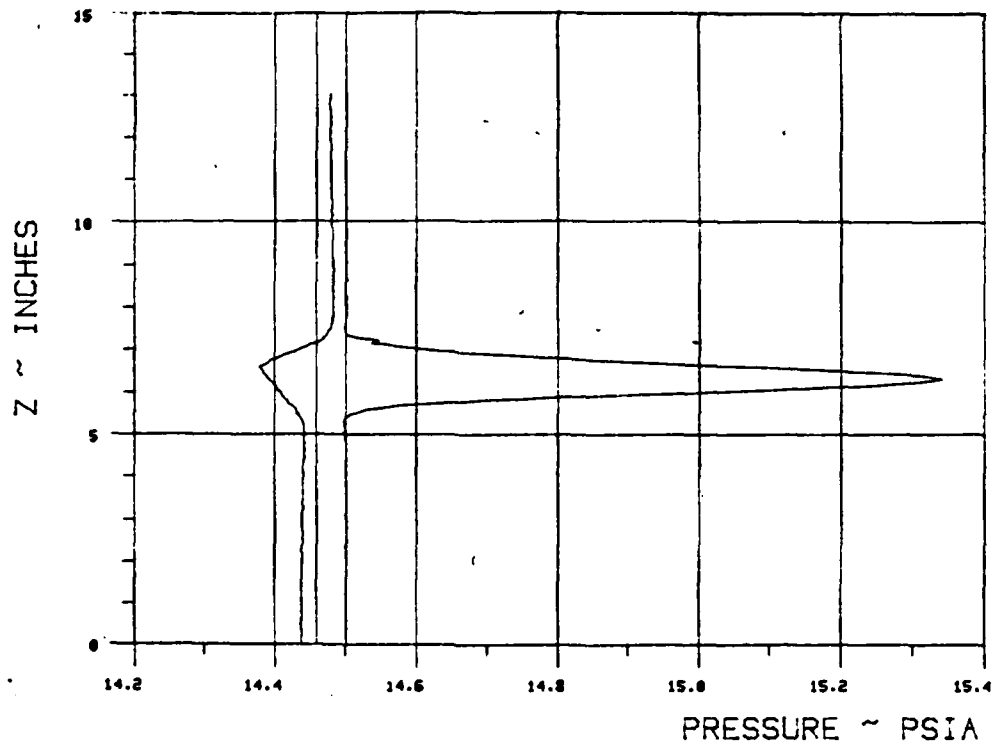


Figure III-13.

WAKE SURVEY DATA

TEST 287 RUN 55

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 4.1113



$X_{ss}/\text{chord} = .739$ $Y_{ss}/\text{chord} = .427$

BETA = 10

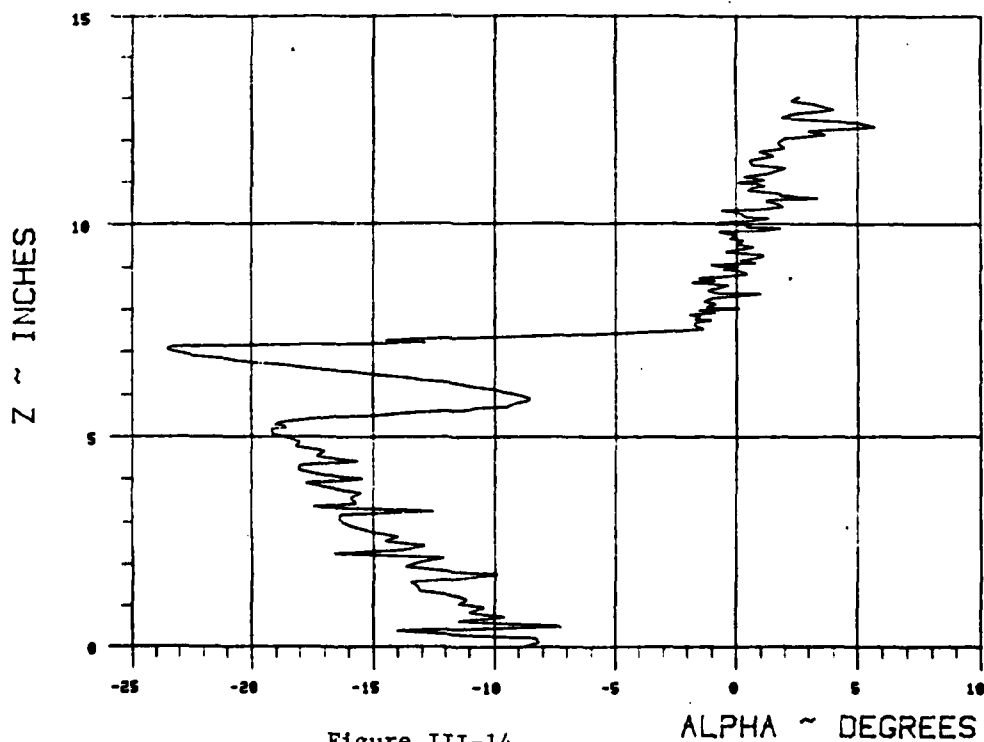
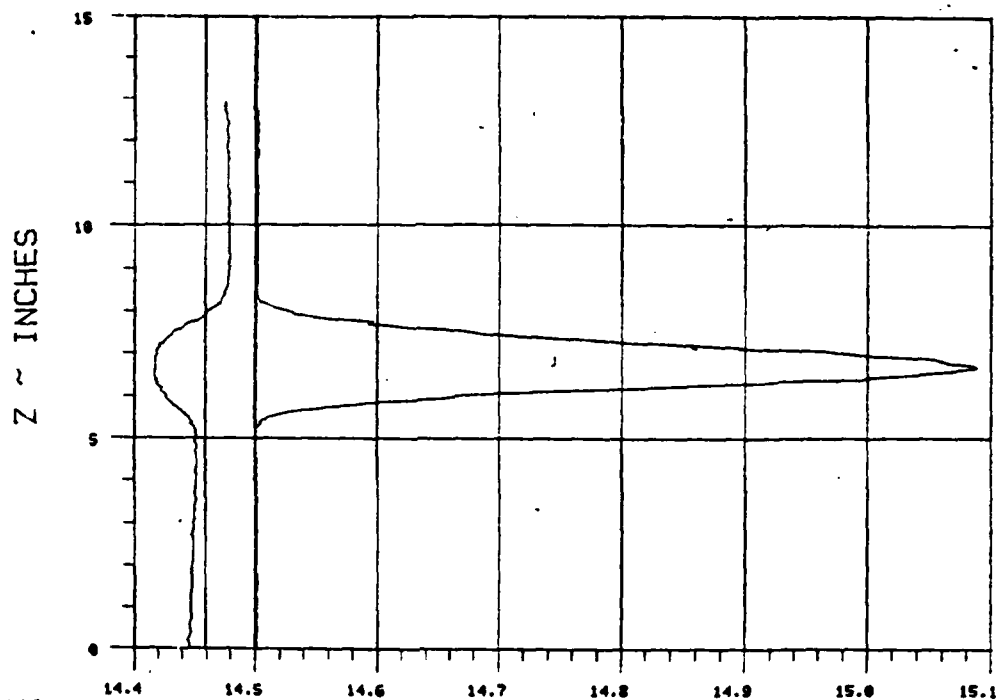


Figure III-14.

WAKE SURVEY DATA

TEST 287 RUN 56

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 4.102



PRESSURE ~ PSIA

$X_{ss}/\text{chord} = \underline{1.108}$ $Y_{ss}/\text{chord} = \underline{.362}$

BETA = 10

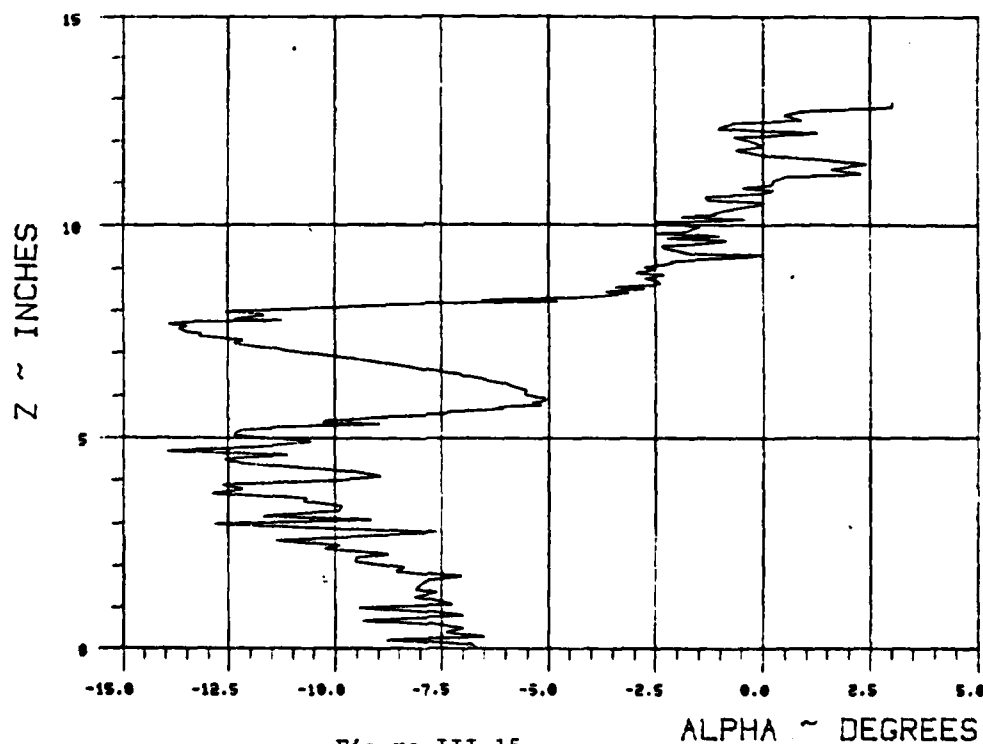


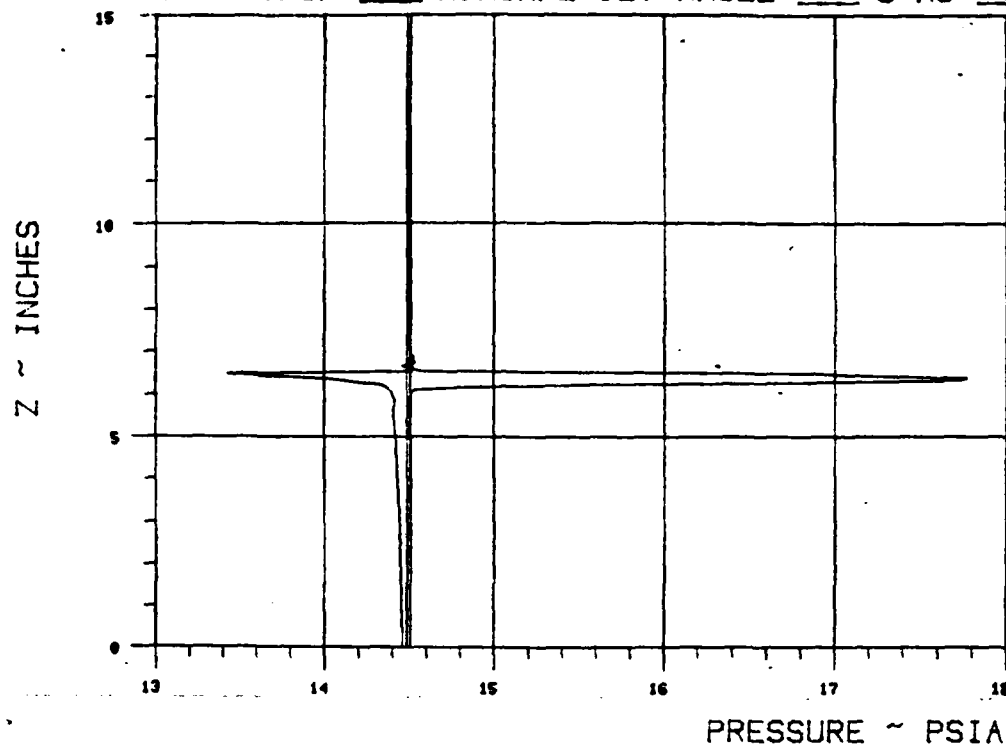
Figure III-15.

ALPHA ~ DEGREES

WAKE SURVEY DATA

TEST 287 RUN 57

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 8.329



$X_{ss}/\text{chord} = .268$ $Y_{ss}/\text{chord} = .637$

BETA = 10

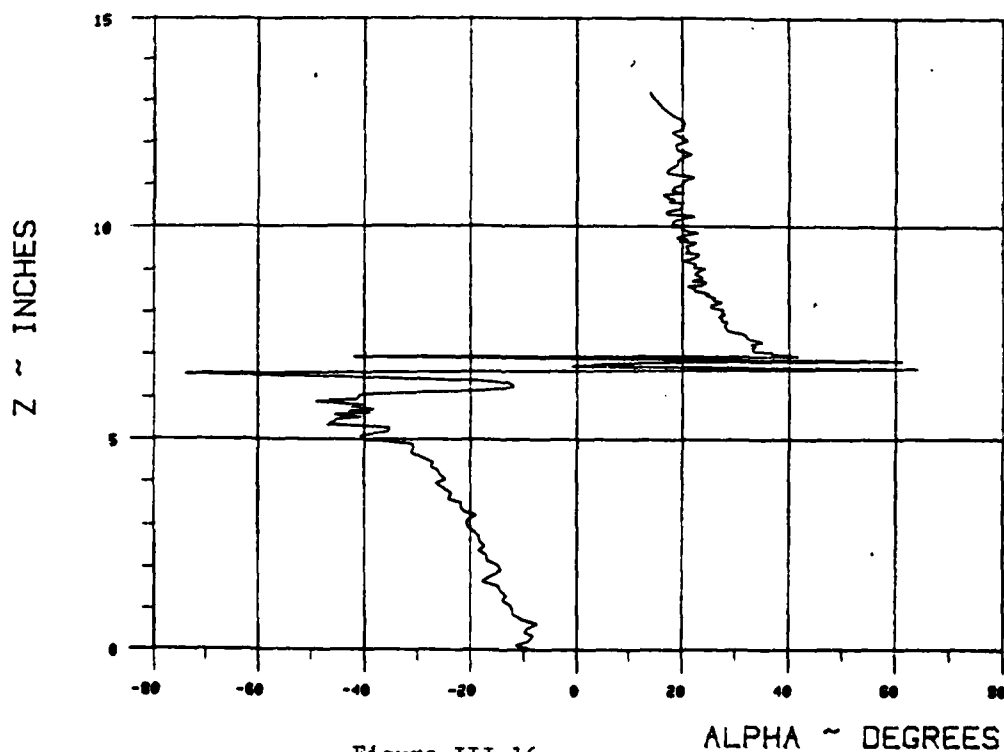
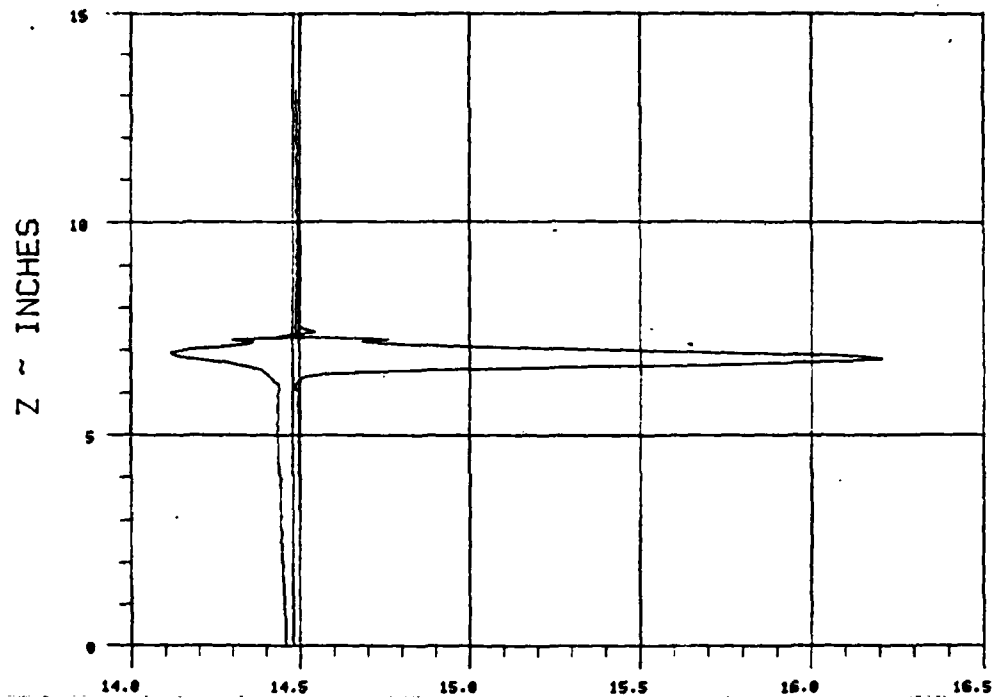


Figure III-16.

WAKE SURVEY DATA

TEST 287 RUN 58

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 8.3877



PRESSURE ~ PSIA

$X_{ss}/\text{chord} = .392$ $Y_{ss}/\text{chord} = .616$

BETA = 10

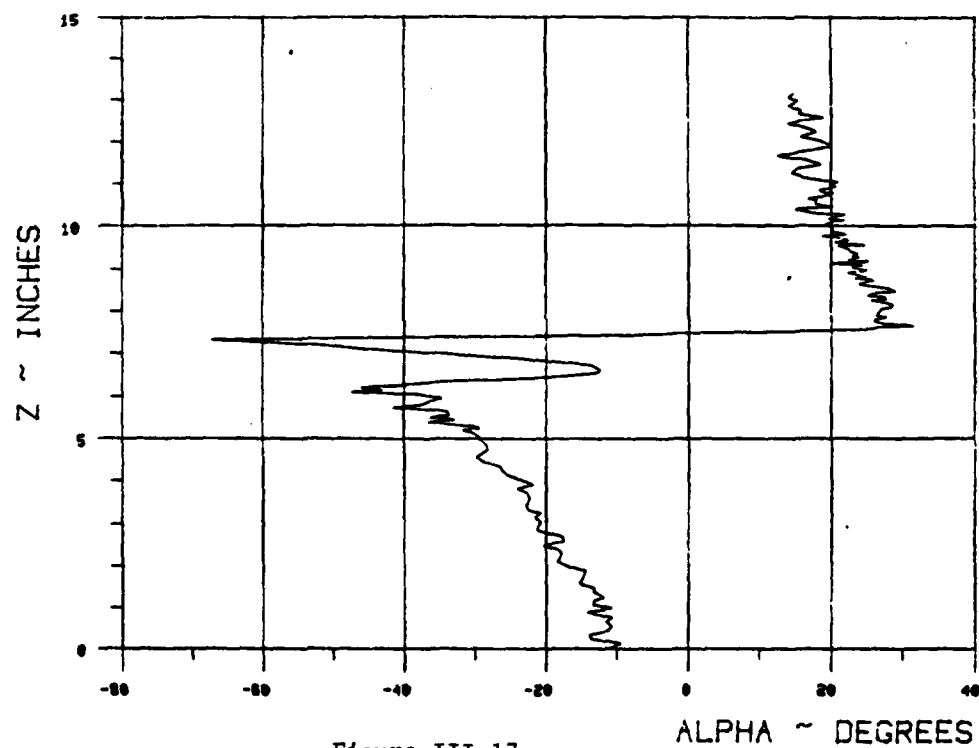
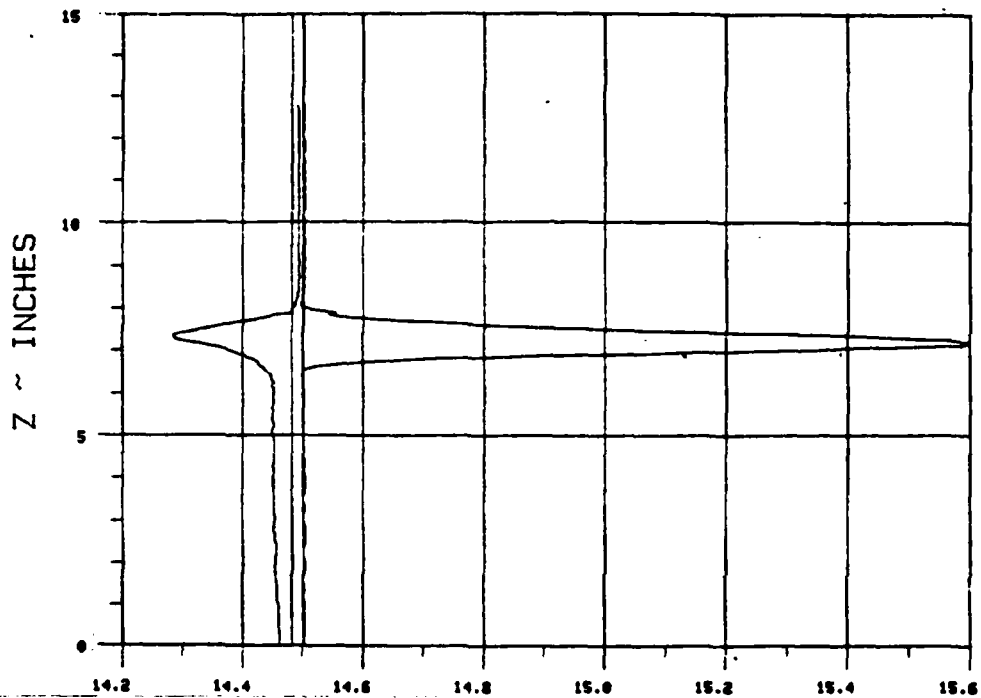


Figure III-17.

WAKE SURVEY DATA

TEST 287 RUN 59

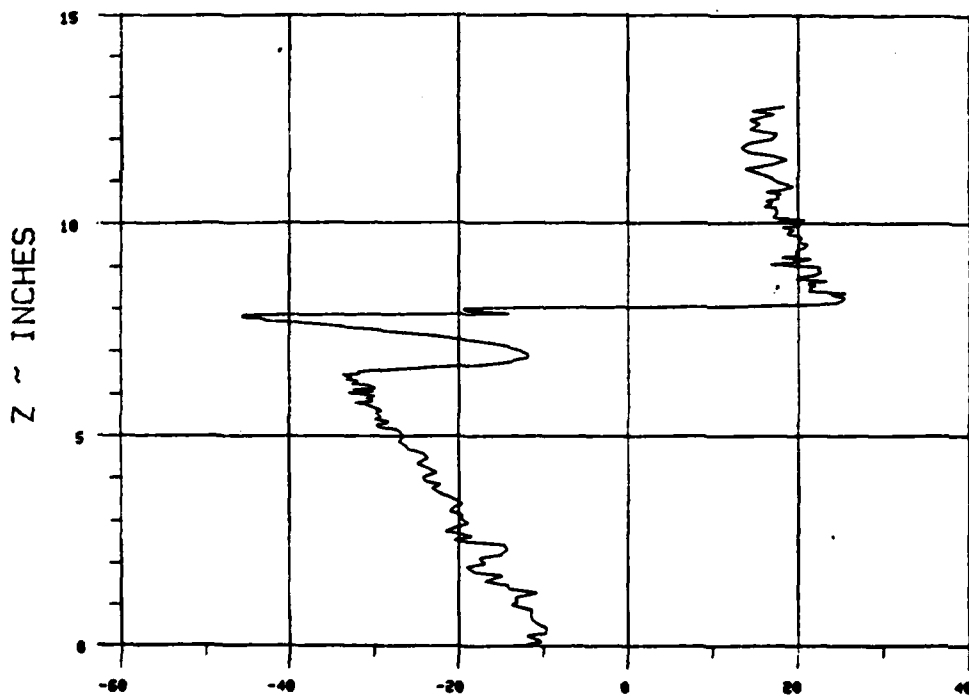
ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 8.6606



PRESSURE ~ PSIA

$X_{ee}/\text{chord} = .515$ $Y_{ee}/\text{chord} = .594$

BETA = 10



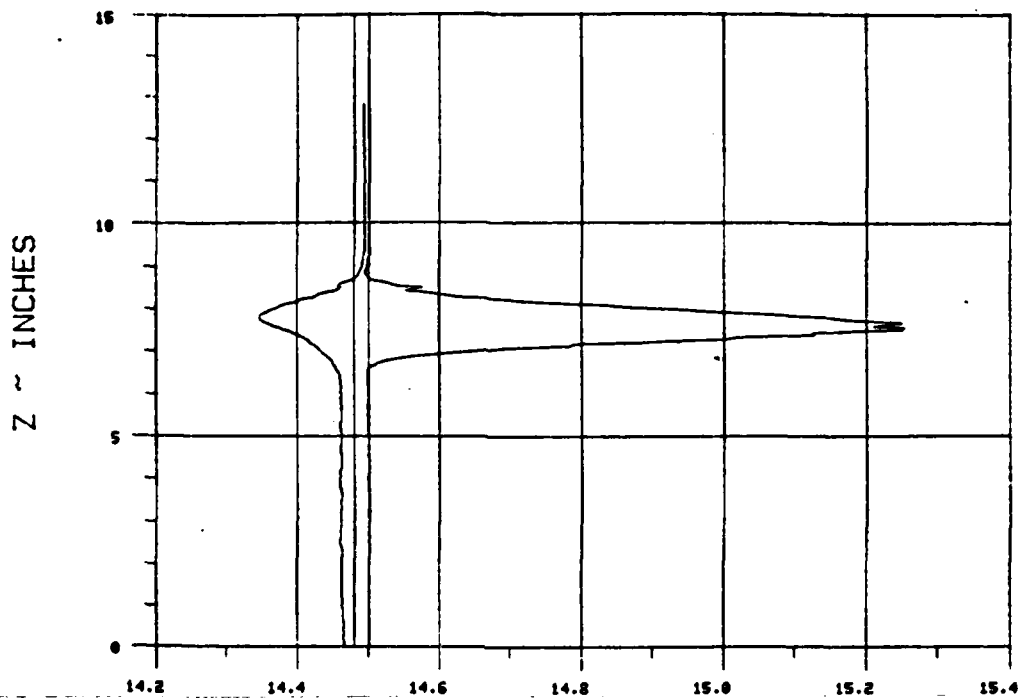
ALPHA ~ DEGREES

Figure III-18.

WAKE SURVEY DATA

TEST 287 RUN 60

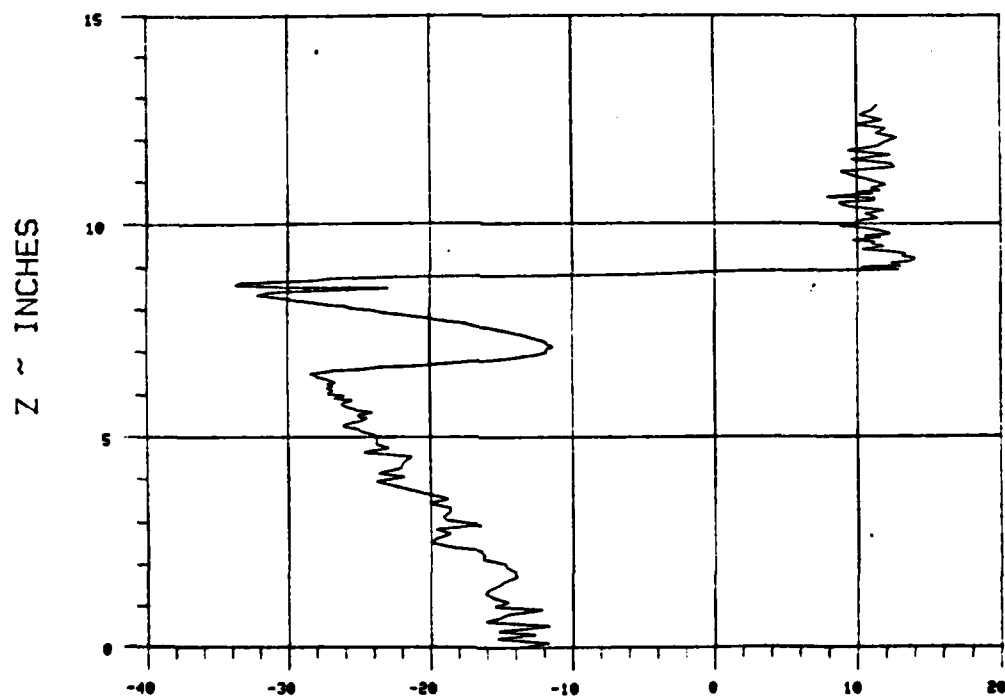
ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MIJ 8.4272



PRESSURE ~ PSIA

$X_{ss}/\text{chord} = .761$ $Y_{ss}/\text{chord} = .55$

BETA = 10



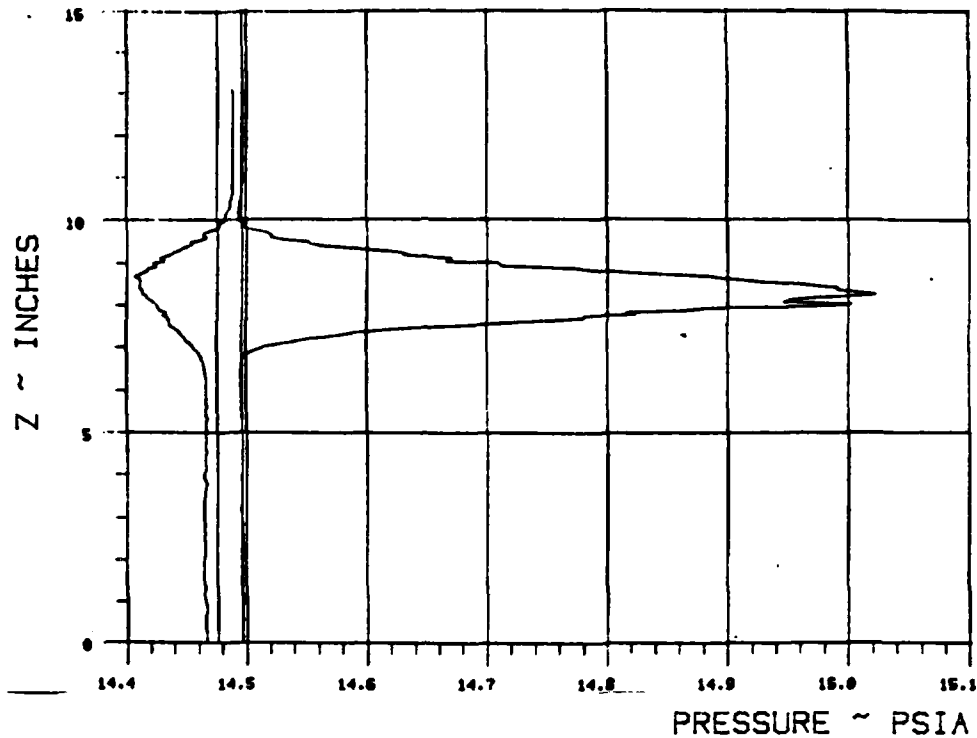
ALPHA ~ DEGREES

Figure III-19.

WAKE SURVEY DATA

TEST 287 RUN 61

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 0.5653



$X_{ss}/\text{chord} = \underline{1.13}$ $Y_{ss}/\text{chord} = \underline{.485}$

BETA = 10

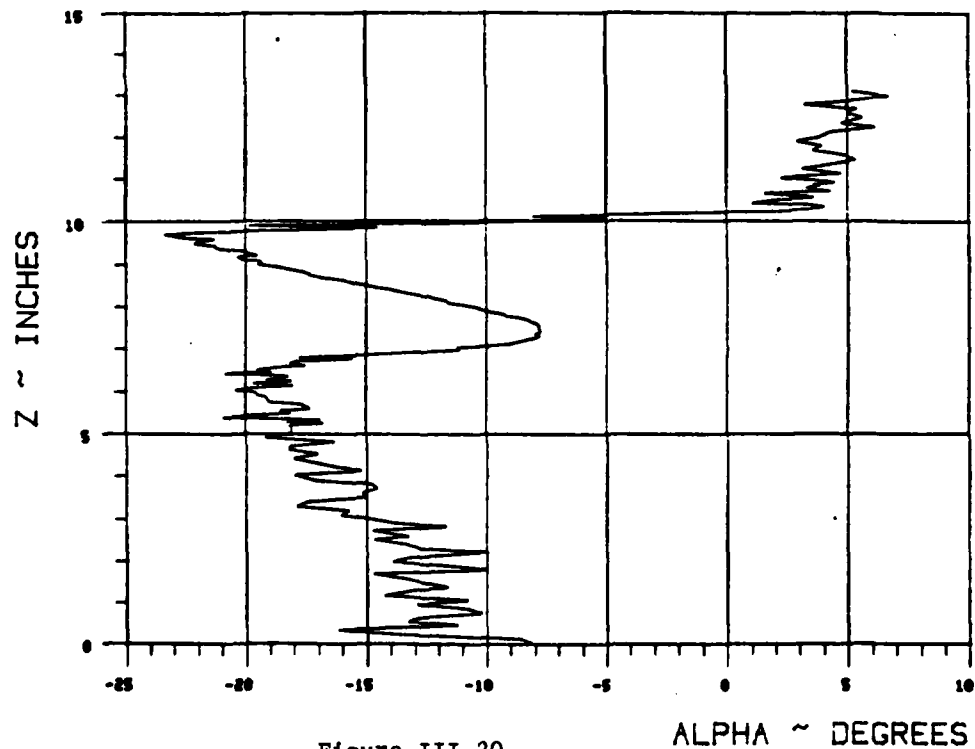
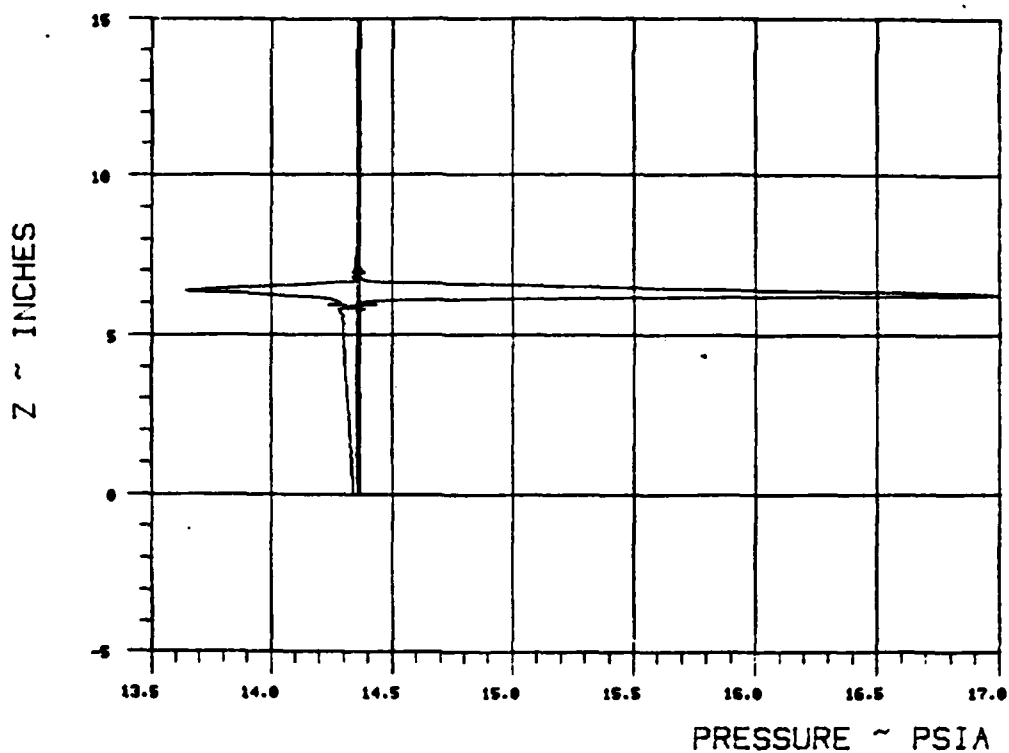


Figure III-20.

WAKE SURVEY DATA

TEST 287 RUN 62

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 17.2768



$X_{ss}/\text{chord} = .268$ $Y_{ss}/\text{chord} = .637$

BETA = 10

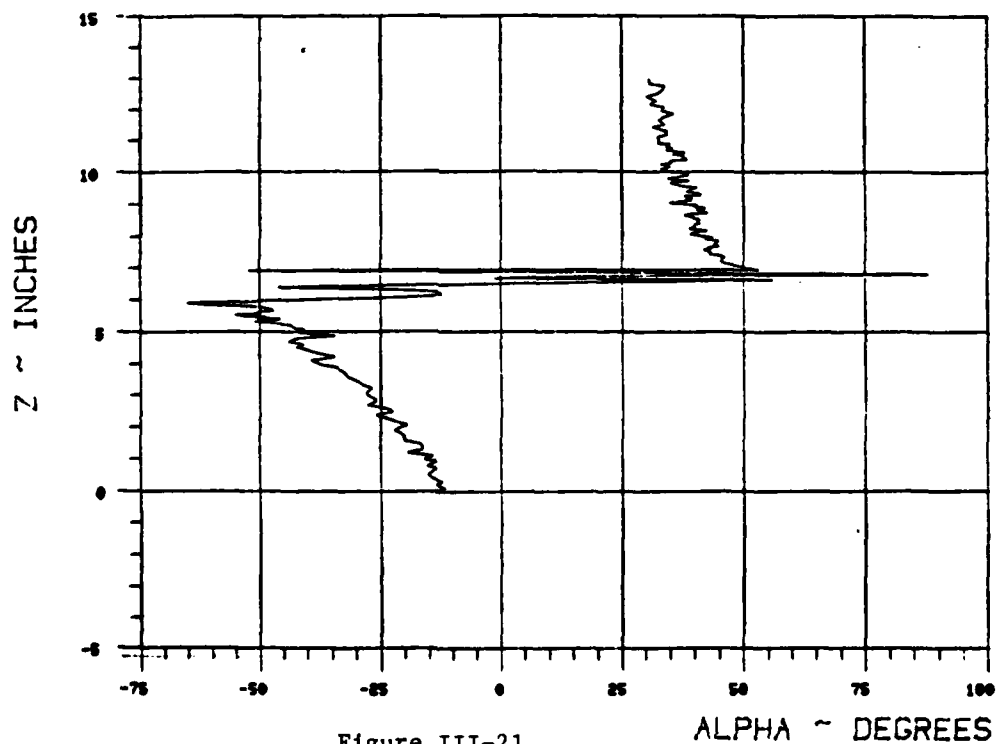
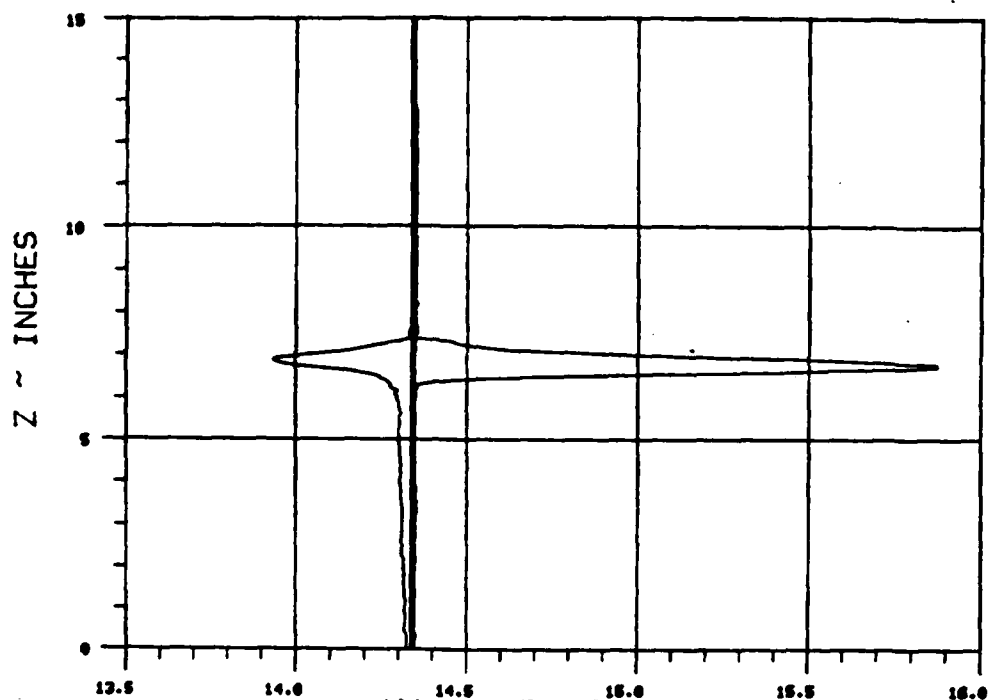


Figure III-21.

WAKE SURVEY DATA

TEST 287 RUN 63

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 16.8459



PRESSURE ~ PSIA

$X_{ee}/\text{chord} = .392$ $Y_{ee}/\text{chord} = .616$

BETA = 10

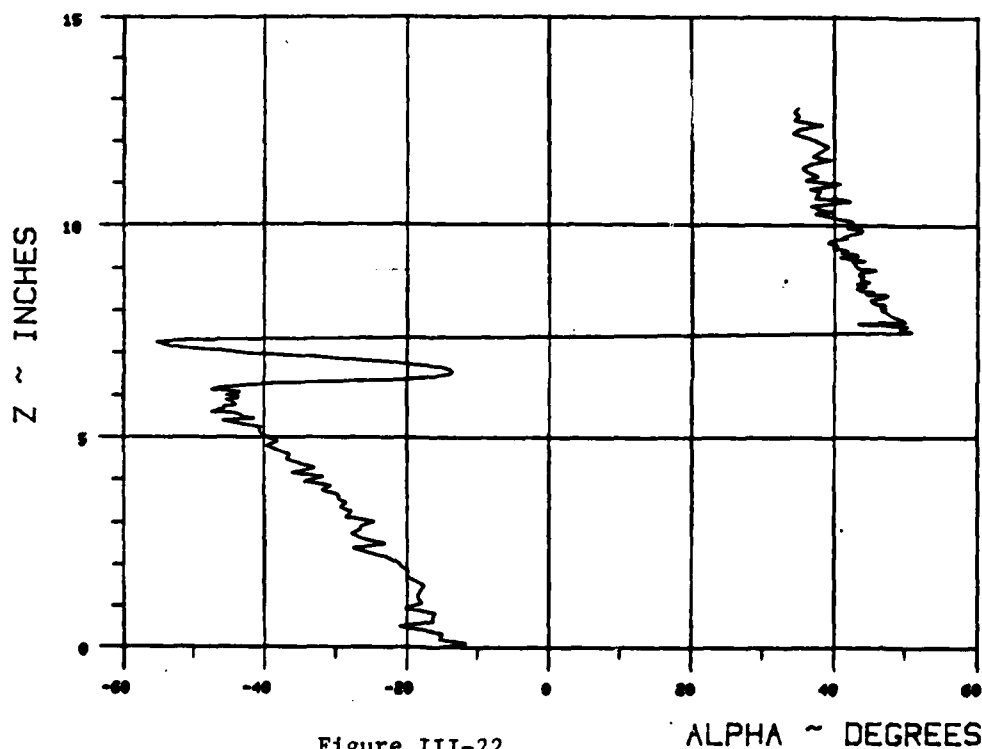
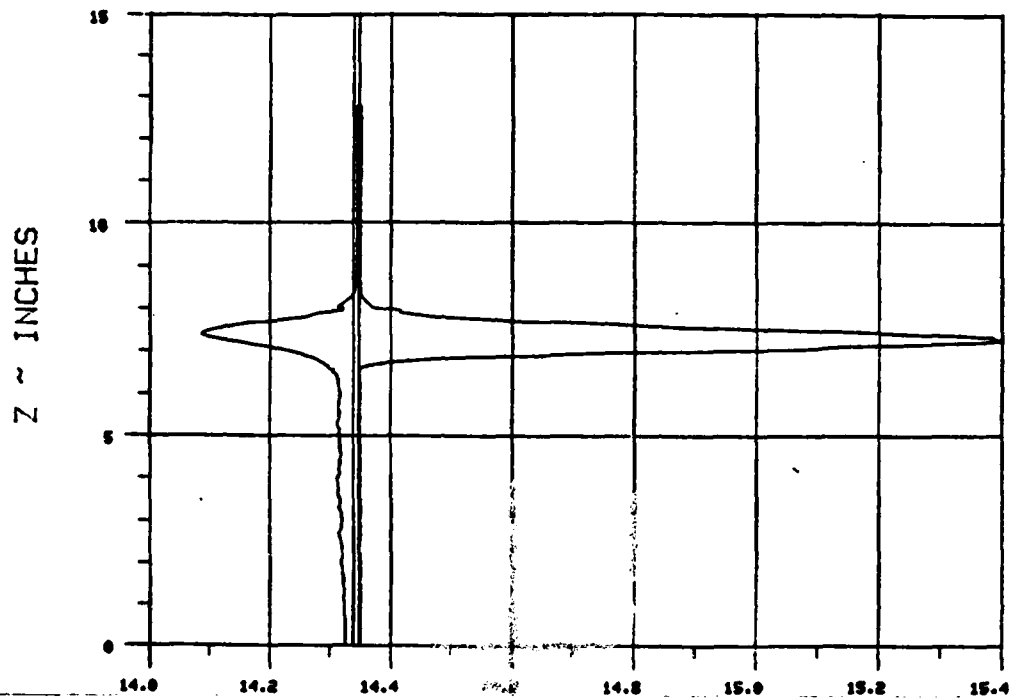


Figure III-22.

WAKE SURVEY DATA

TEST 287 RUN 64

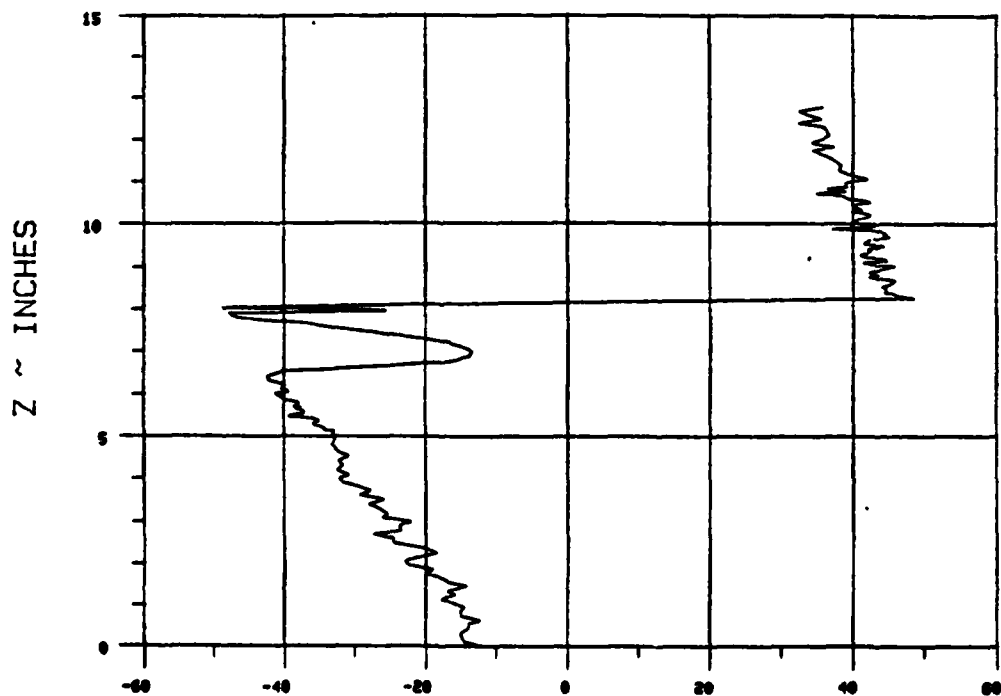
ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 17.3211



PRESSURE ~ PSIA

$X_{se}/\text{chord} = .515$ $Y_{se}/\text{chord} = .594$

BETA = 10



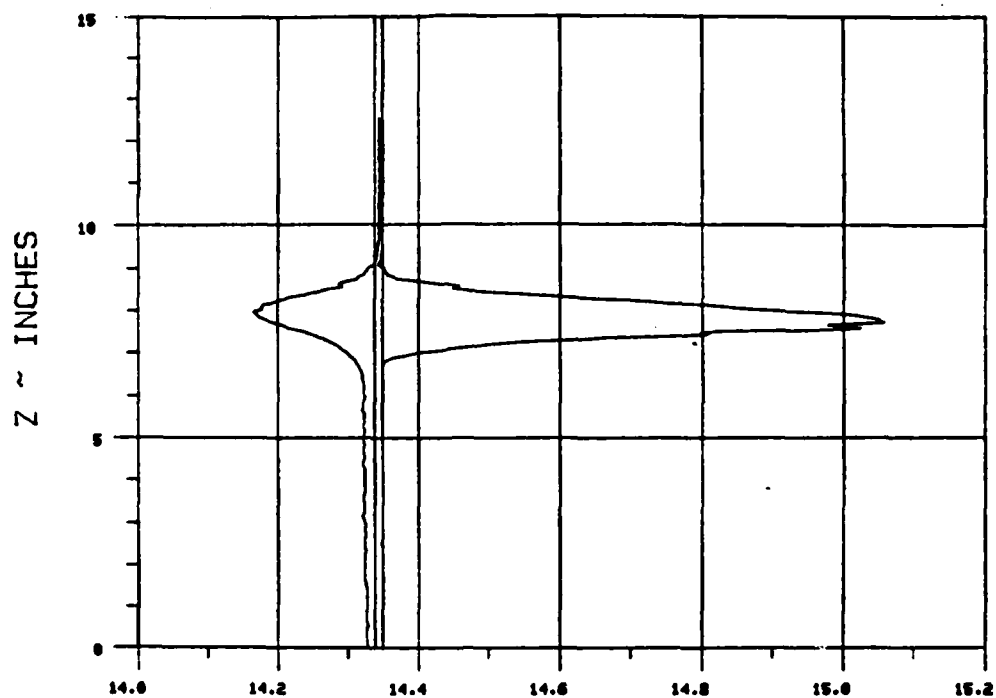
ALPHA ~ DEGREES

Figure III-23.

WAKE SURVEY DATA

TEST 287 RUN 65

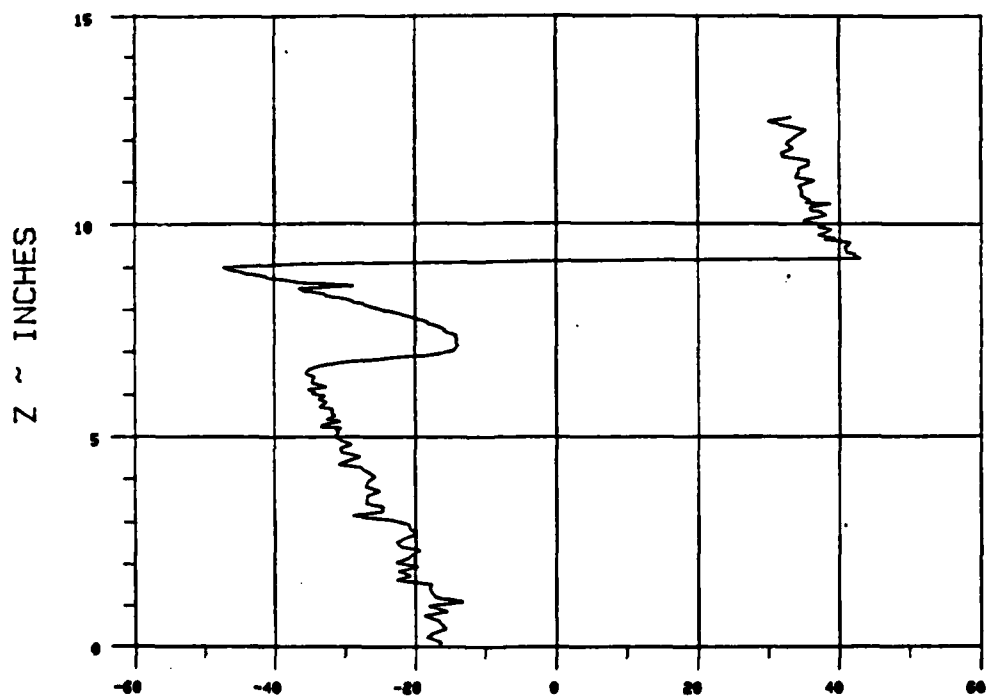
ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 16.9019



PRESSURE ~ PSIA

$X_{se}/chord = .761$ $Y_{se}/chord = .55$

BETA = 10



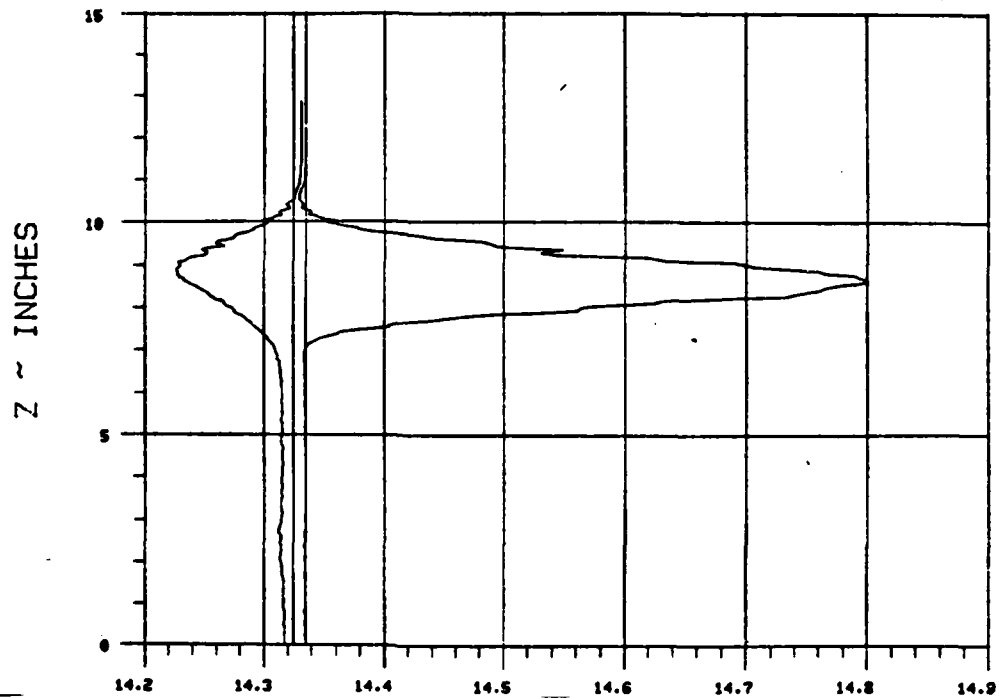
ALPHA ~ DEGREES

Figure III-24.

WAKE SURVEY DATA

TEST 287 RUN 66

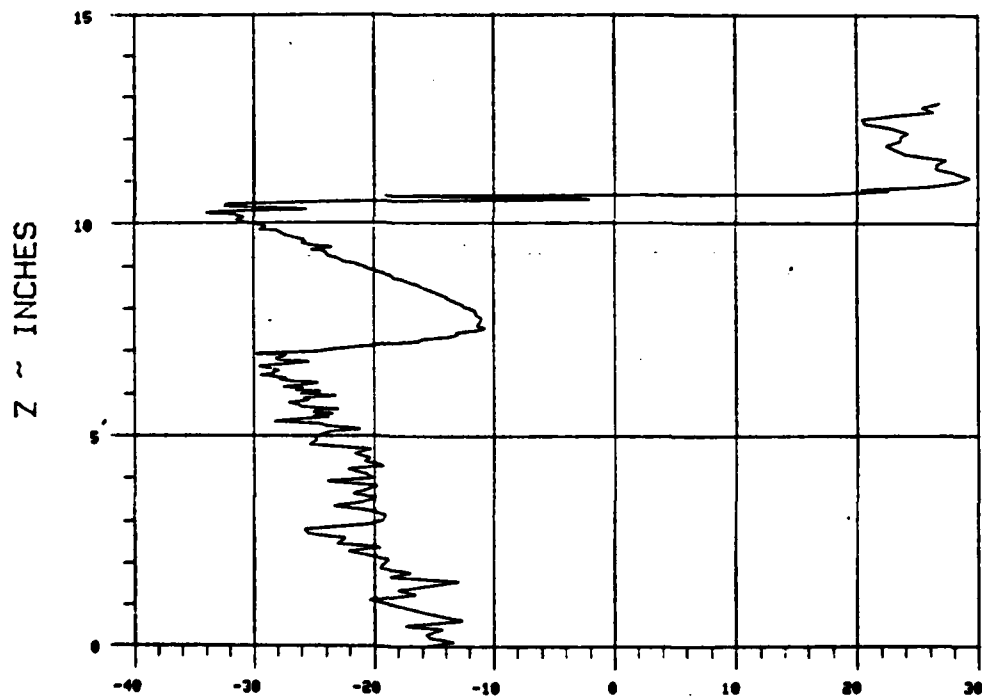
ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 17.6394



PRESSURE ~ PSIA

$X_{ss}/\text{chord} = \underline{1.13}$ $Y_{ss}/\text{chord} = \underline{.485}$

BETA = 10



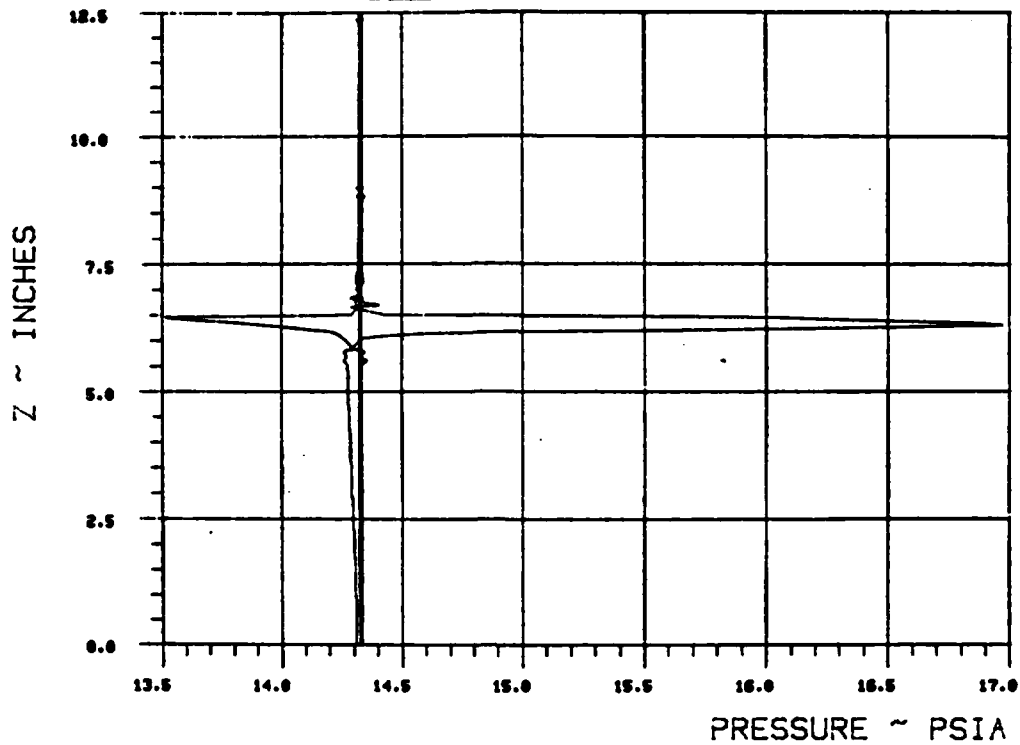
ALPHA ~ DEGREES

Figure III-25.

WAKE SURVEY DATA

TEST 287 RUN 67

ANGLE OF ATTACK 0 NOMINAL JET. ANGLE 30 C MU 25.4067



$X_{ss}/\text{chord} = .268$ $Y_{ss}/\text{chord} = .637$

BETA = 10

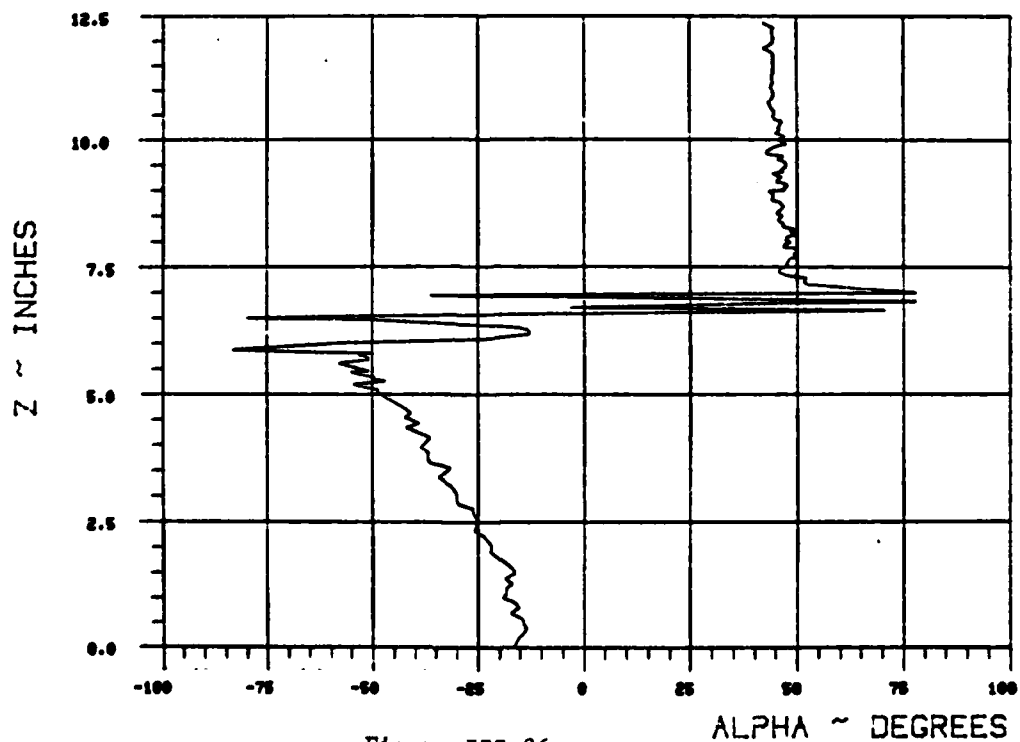
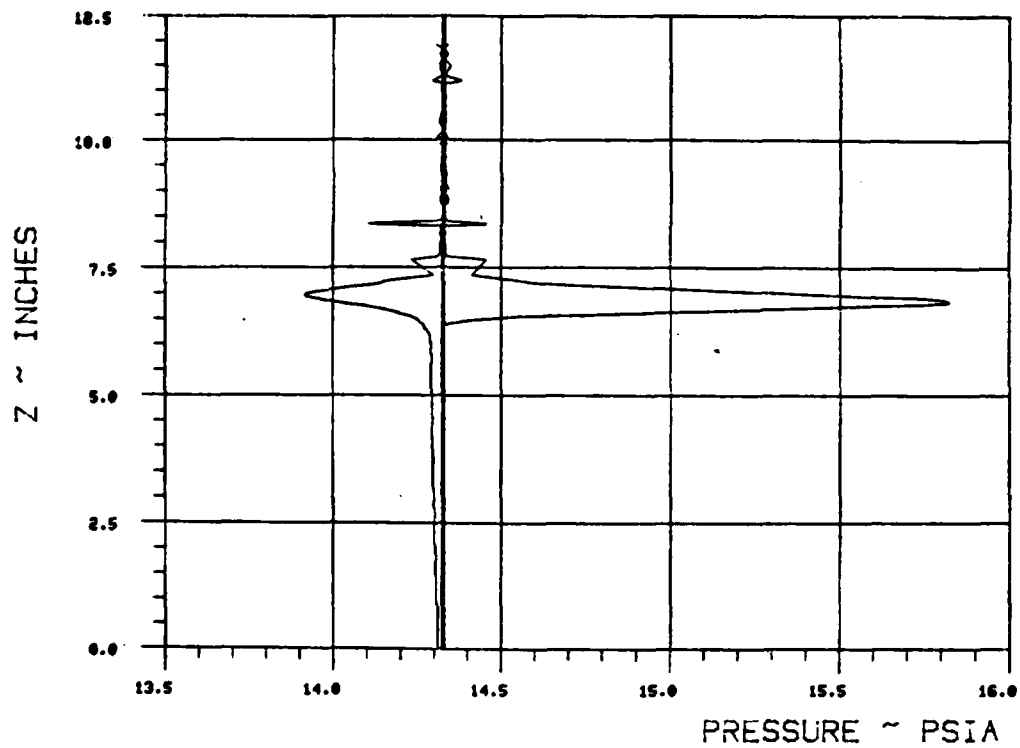


Figure III-26.

WAKE SURVEY DATA

TEST 207 RUN 68

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 26.77/3



$X_{ss}/\text{chord} = .392$ $Y_{ss}/\text{chord} = .616$

BETA = 10

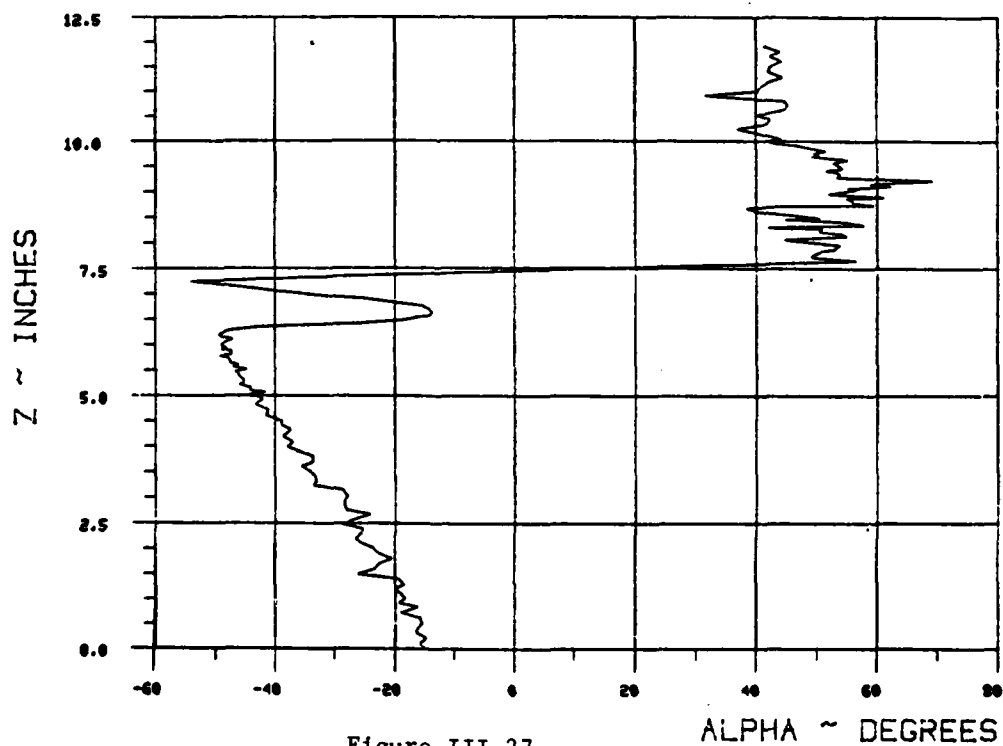
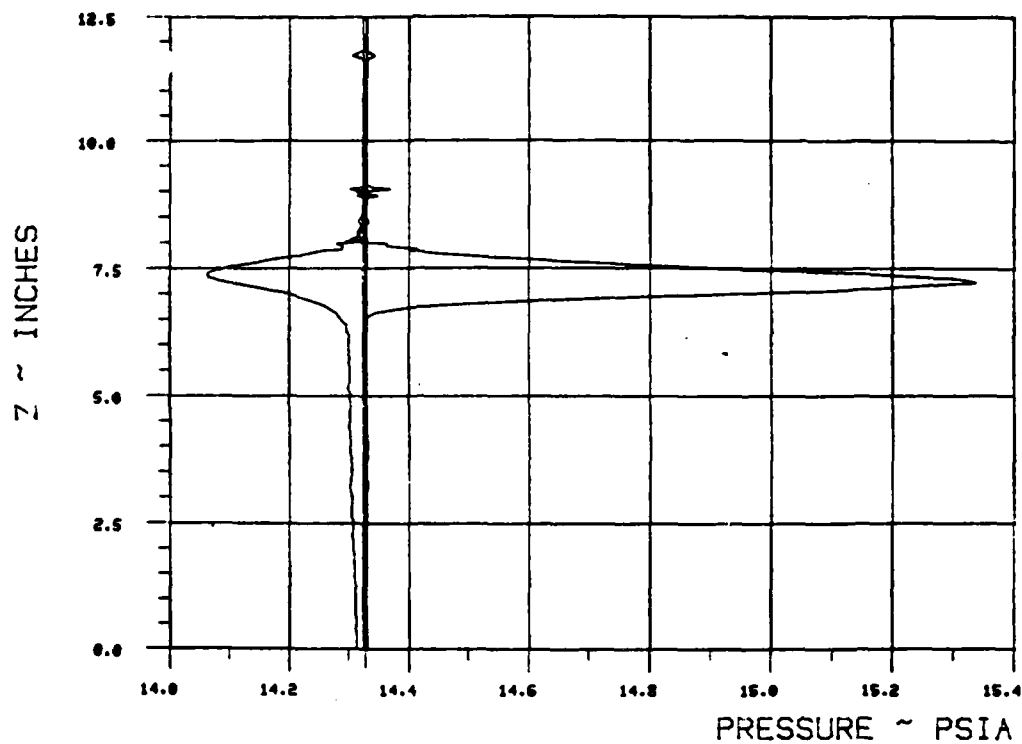


Figure III-27.

WAKE SURVEY DATA

TEST 287 RUN 69

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 25.8256



$X_{ss}/\text{chord} = .515$ $Y_{ss}/\text{chord} = .594$

BETA = 10

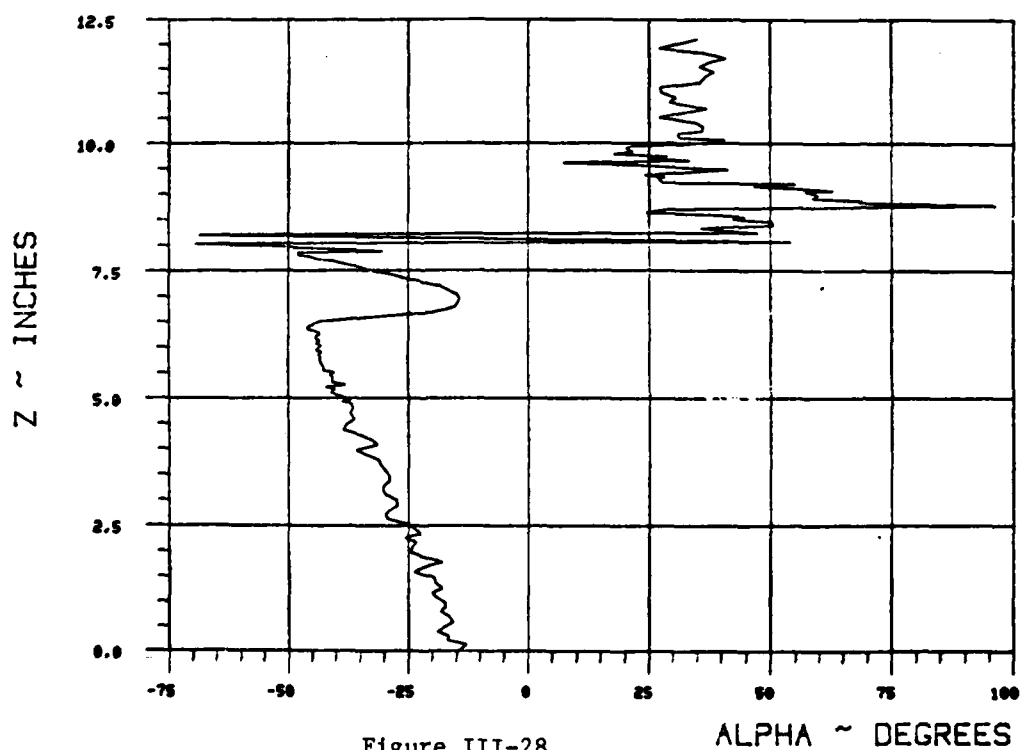
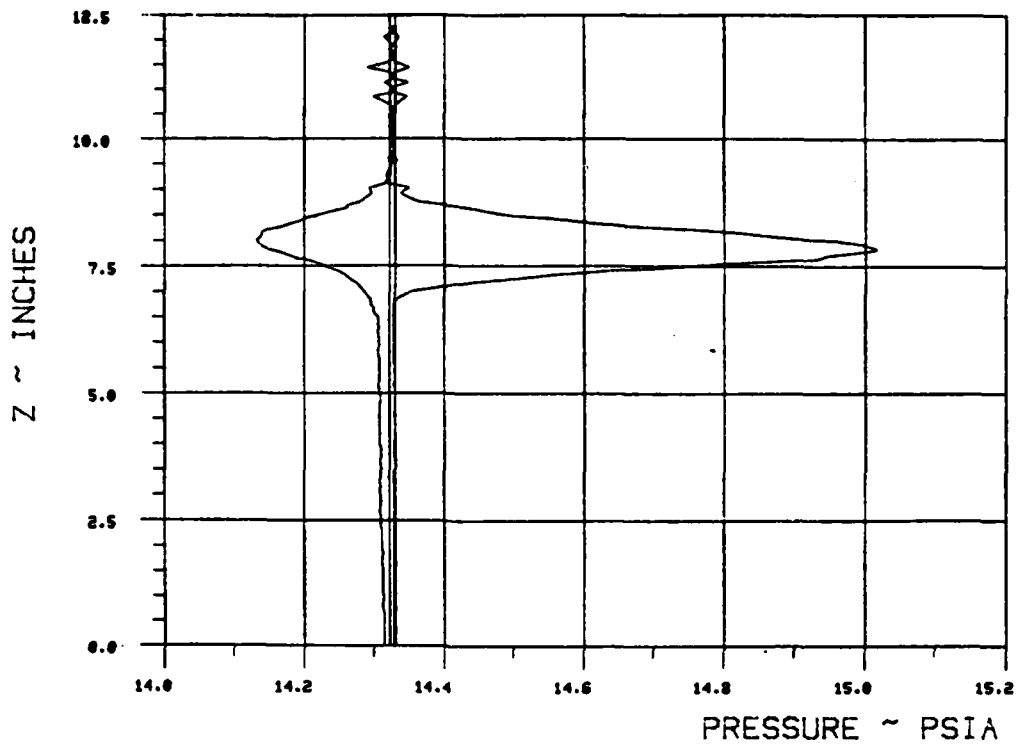


Figure III-28.

WAKE SURVEY DATA

TEST 287 RUN 10

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 24.9456



$X_{ss}/\text{chord} = .761$ $Y_{ss}/\text{chord} = .55$

BETA = 10

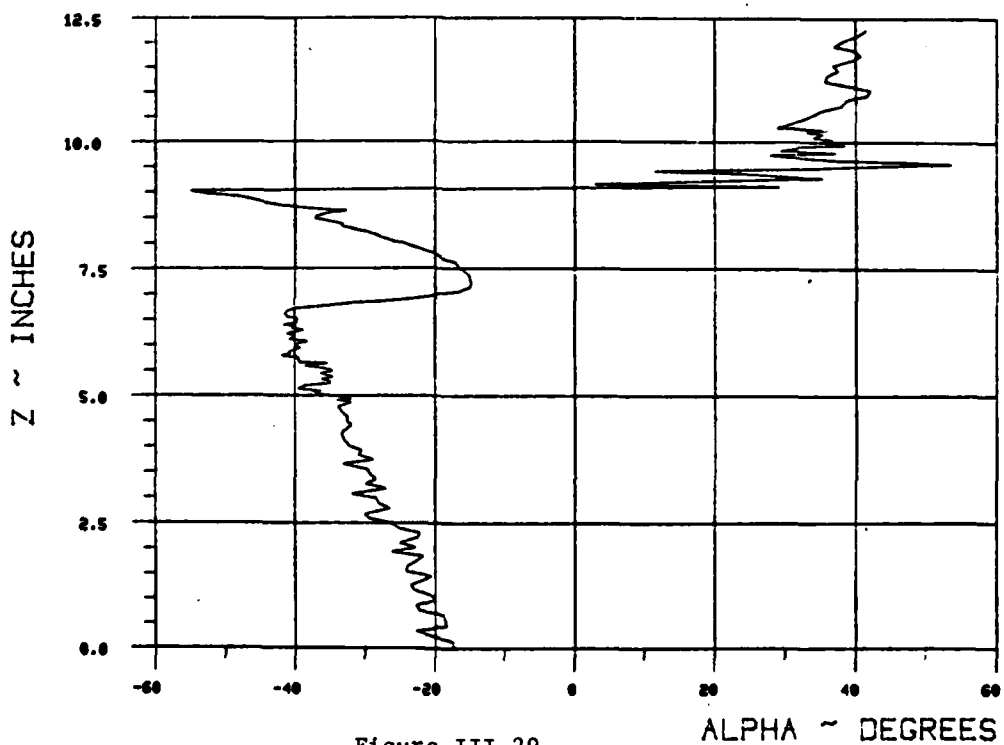
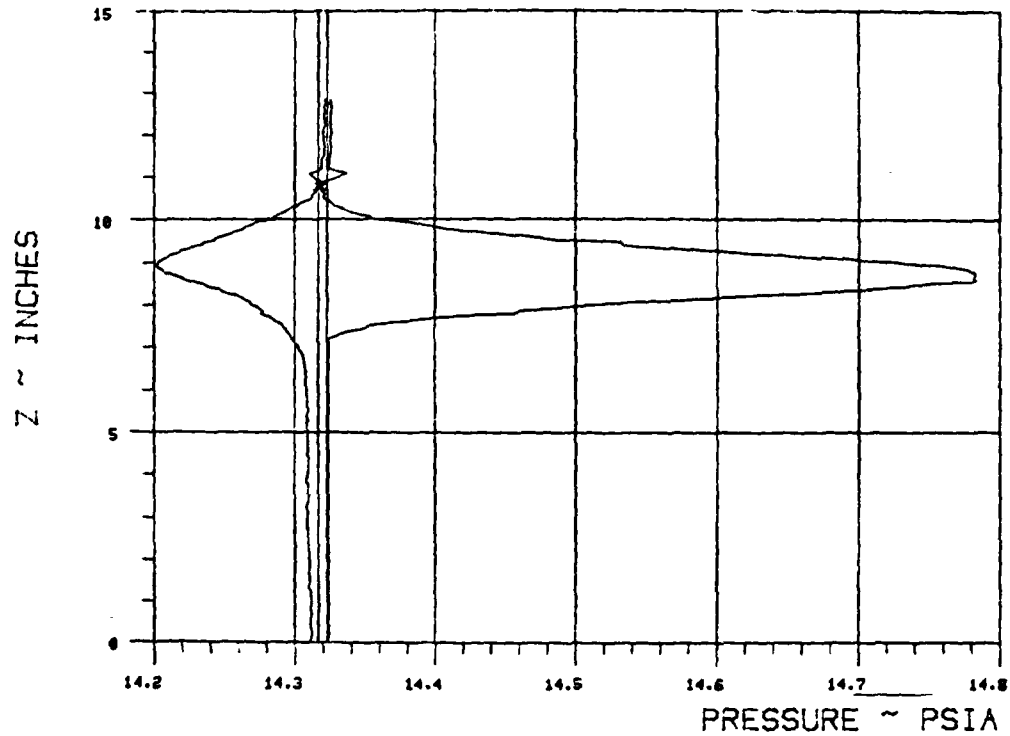


Figure III-29.

WAKE SURVEY DATA

TEST 287 RUN 71

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 30 C MU 26.8944



Xes/chord = 1.13 Yes/chord = .485

BETA = 10

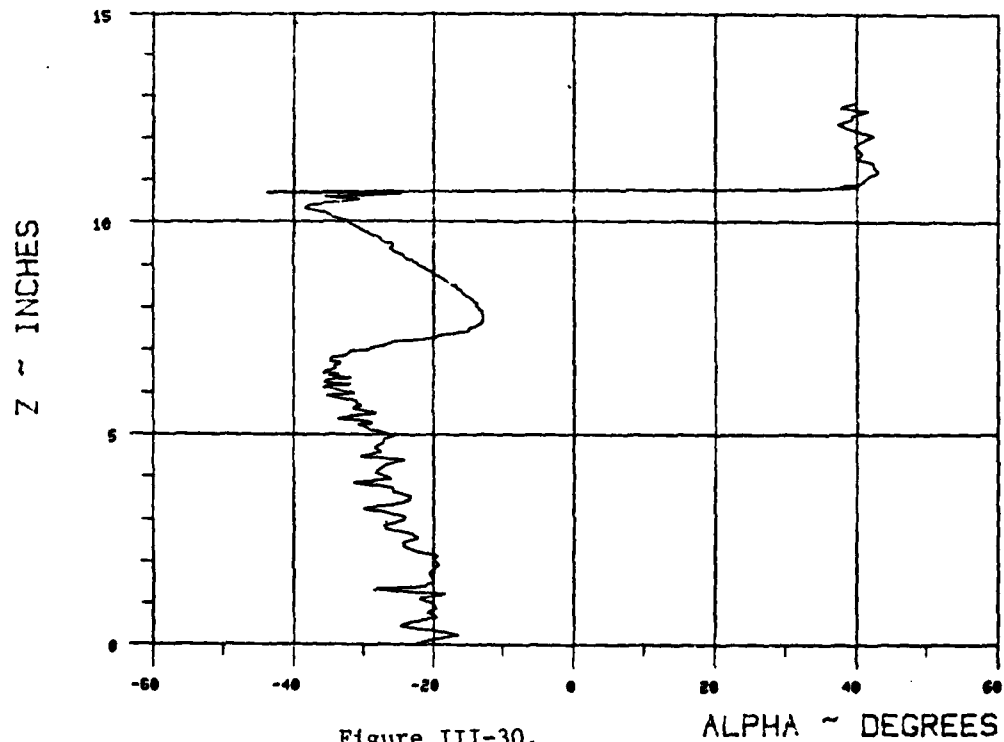
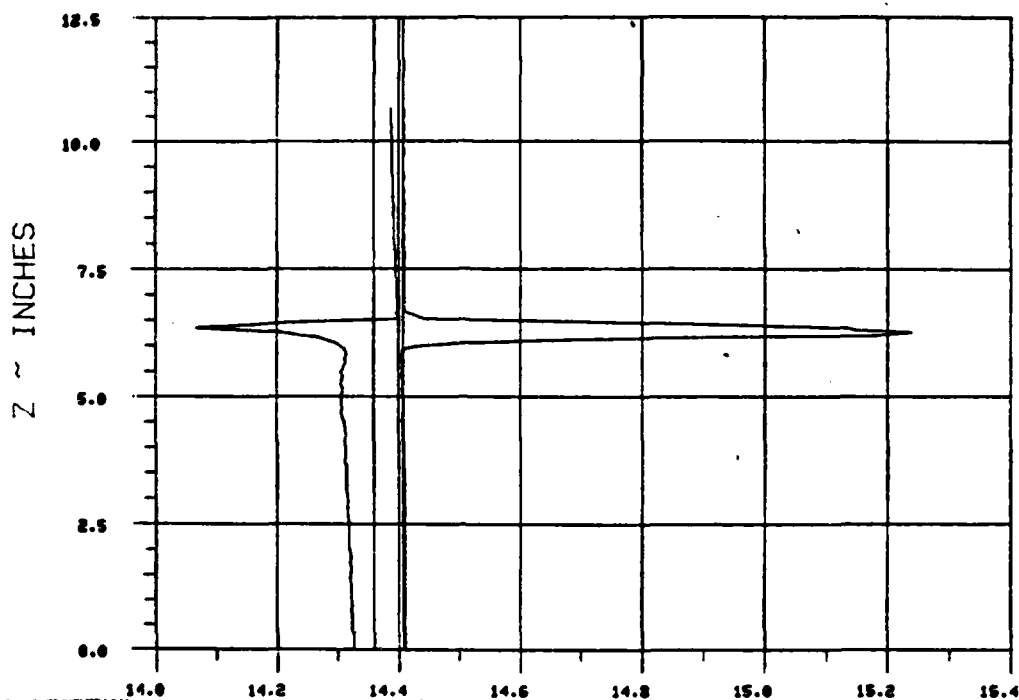


Figure III-30.

WAKE SURVEY DATA

TEST 287 RUN 36

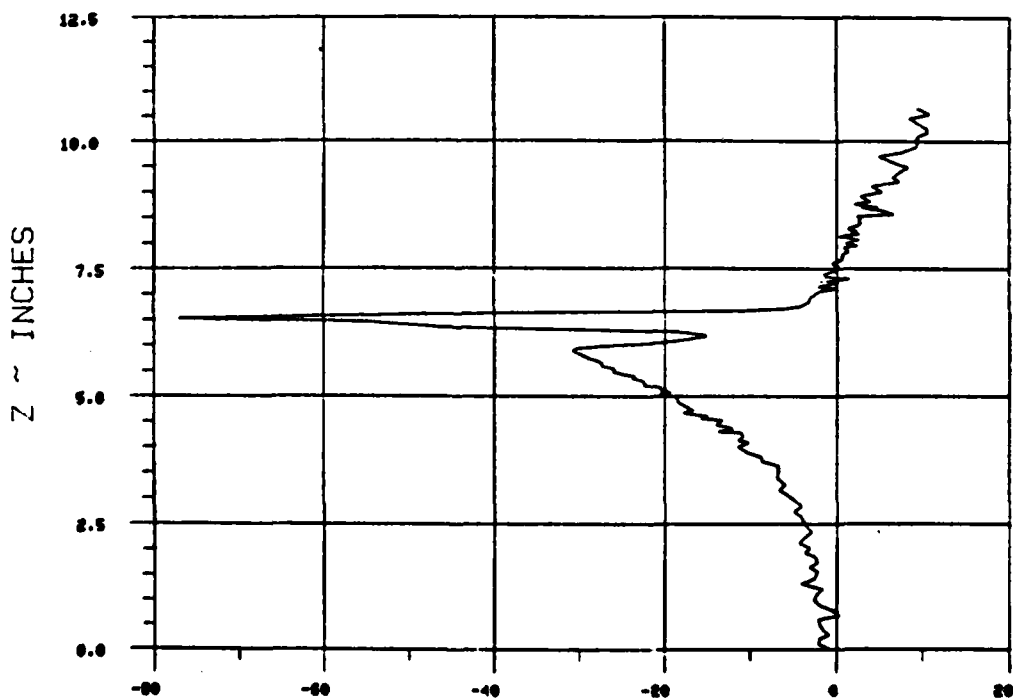
ANGLE OF ATTACK 0 NOMINAL JET ANGLE 50 C MU .9947



PRESSURE ~ PSIA

$X_{ss}/\text{chord} = .368$ $Y_{ss}/\text{chord} = .565$

BETA = 20



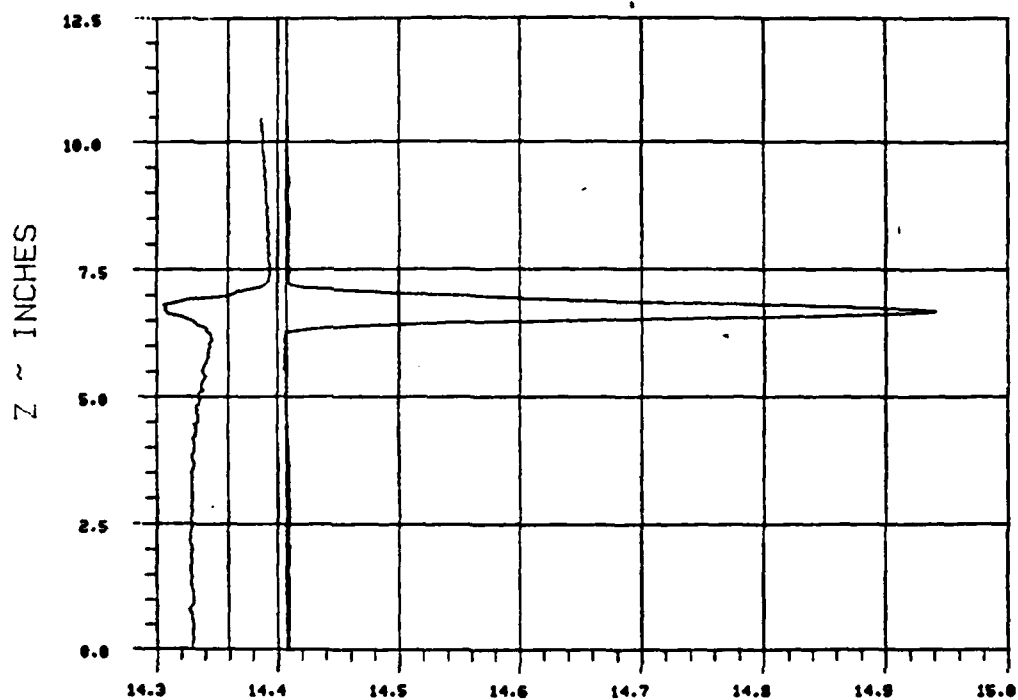
ALPHA ~ DEGREES

Figure III-31.

WAKE SURVEY DATA

TEST 287 RUN 37

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 50 C MU 1.0013



PRESSURE ~ PSIA

$X_{ss}/\text{chord} = .485$ $Y_{ss}/\text{chord} = .522$

BETA = 20

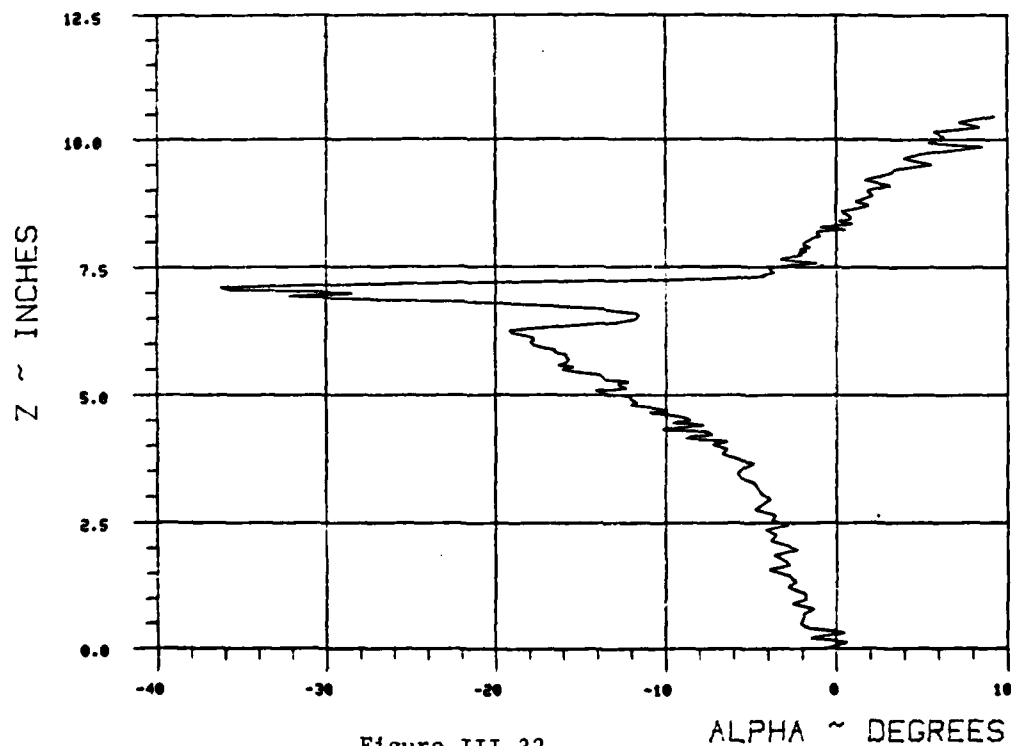
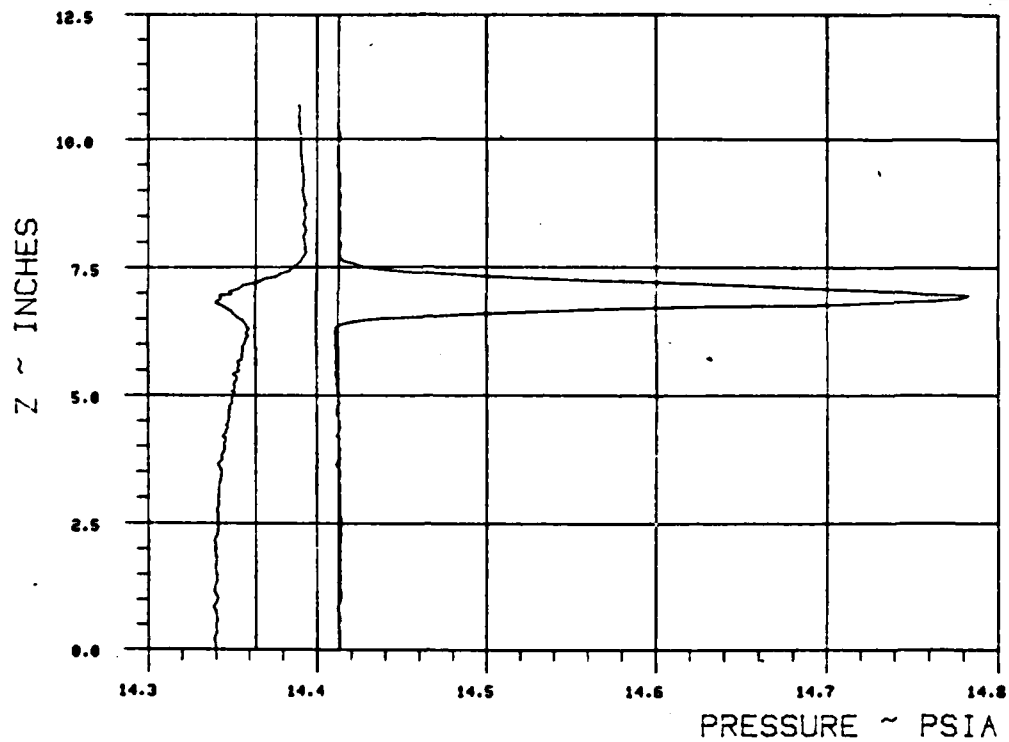


Figure III-32.

WAKE SURVEY DATA

TEST 287 RUN 38

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 50 C MU 1.0041



$X_{ss}/\text{chord} = .602$ $Y_{ss}/\text{chord} = .479$

BETA = 20

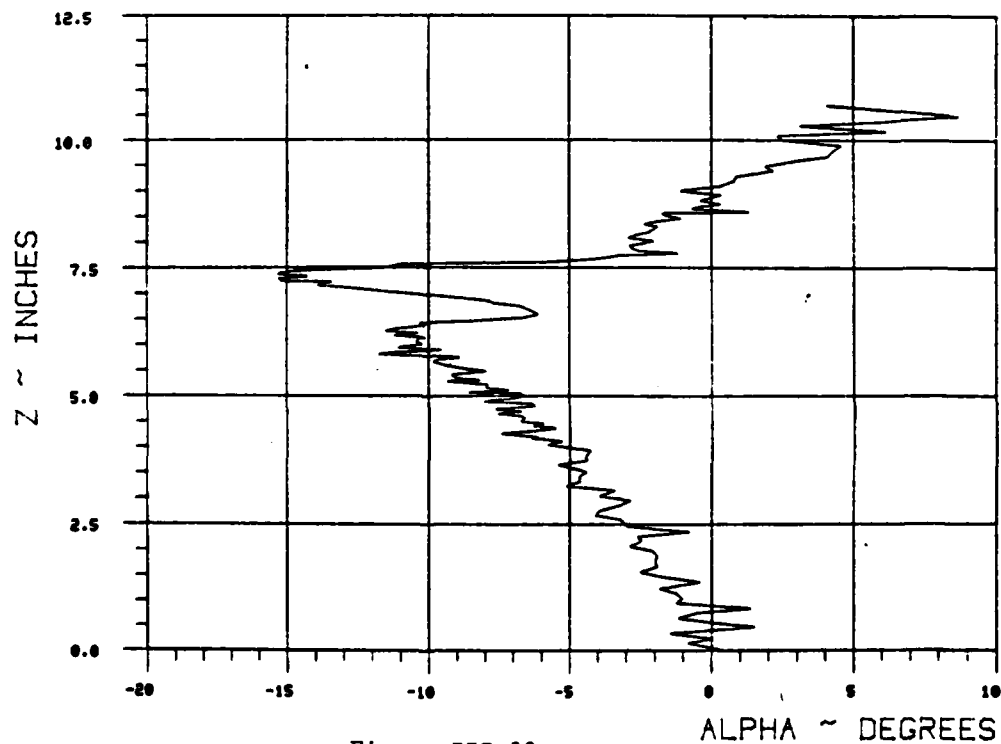
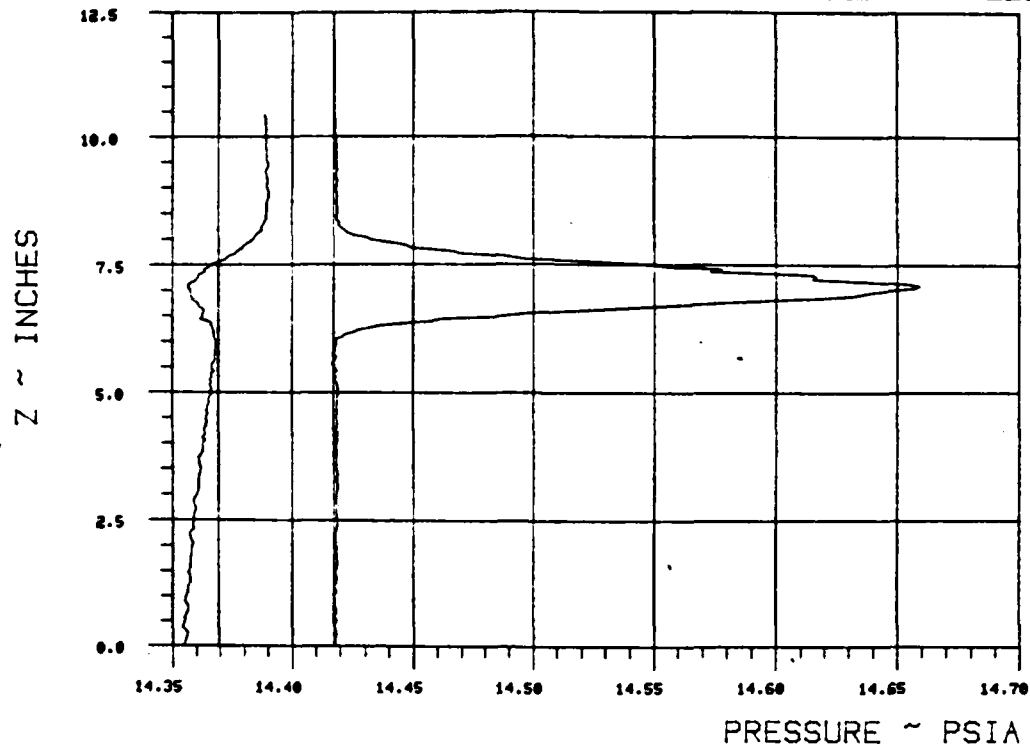


Figure III-33.

WAKE SURVEY DATA

TEST 287 RUN 41

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 50 C MU 1.011



$X_{ss}/\text{chord} = .837$ $Y_{ss}/\text{chord} = .394$

BETA = 20

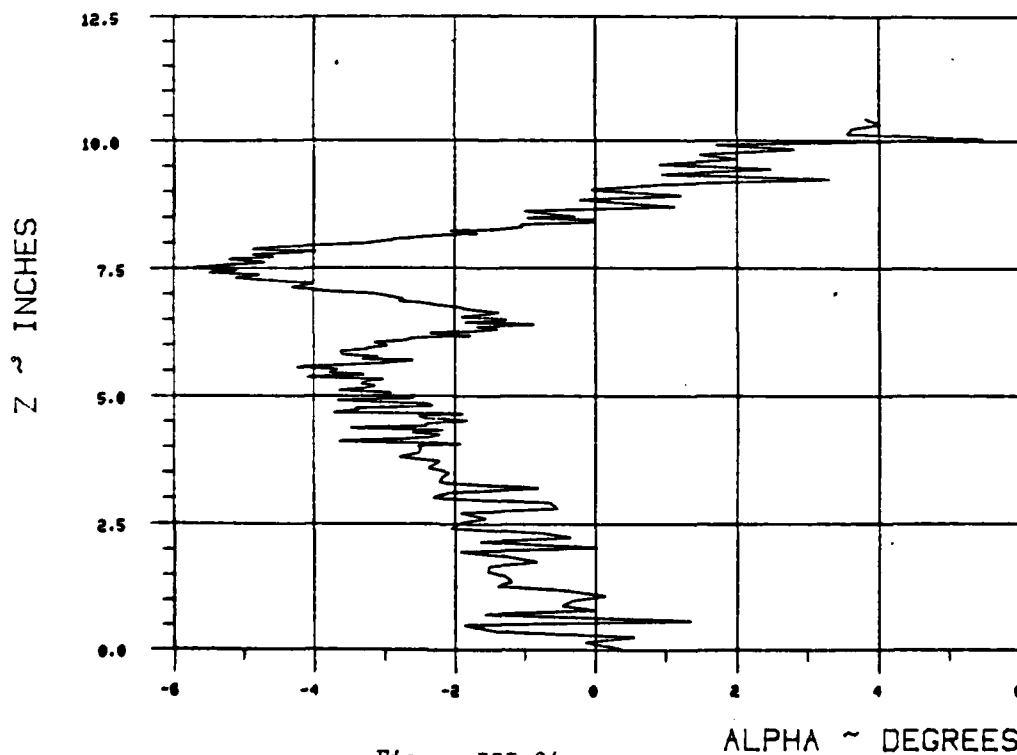
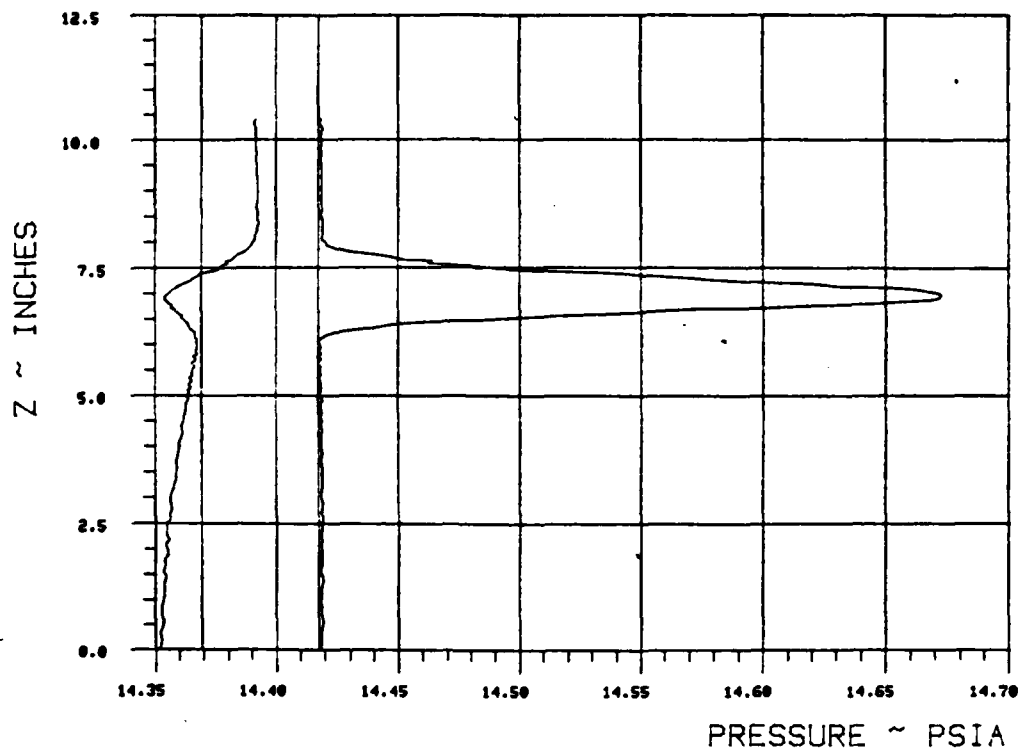


Figure III-34.

WAKE SURVEY DATA

TEST 287 RUN 39

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 50 C MU .9893



$X_{ss}/\text{chord} = .837$ $Y_{ss}/\text{chord} = .397$

BETA = 20

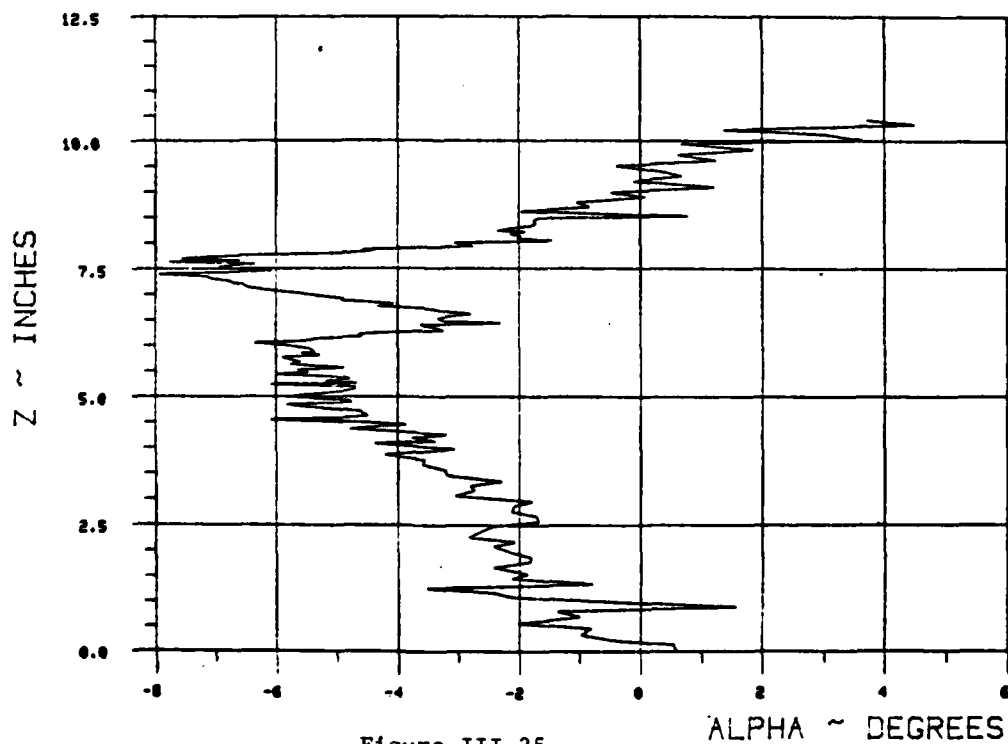
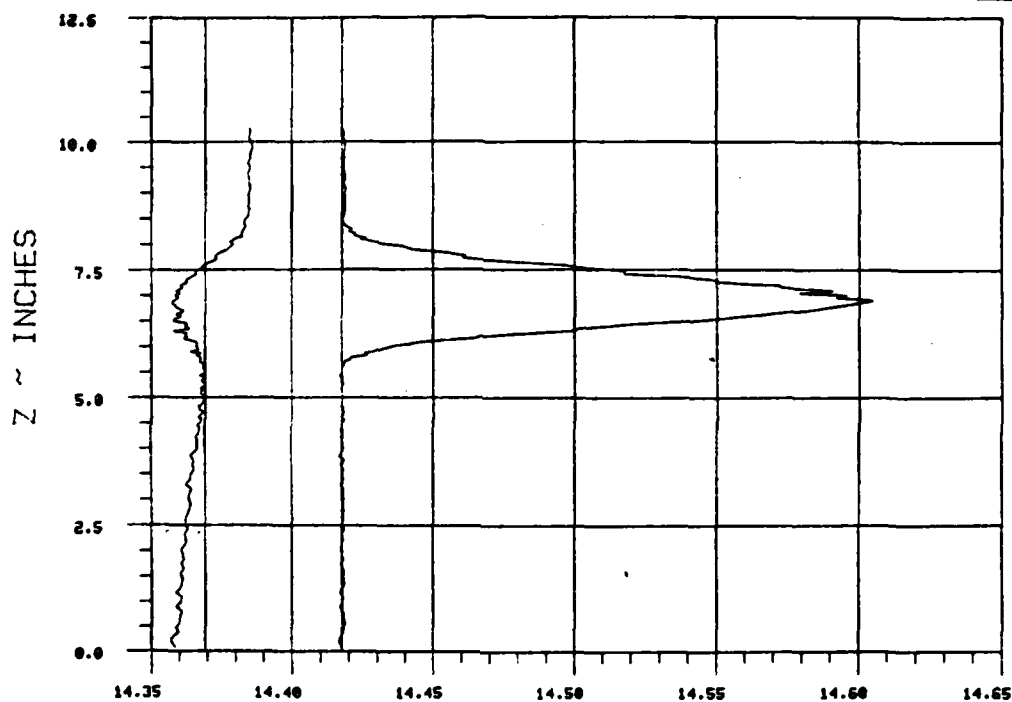


Figure III-35.

WAKE SURVEY DATA

TEST 287 RUN 40

ANGLE OF ATTACK 0 NOMINAL JET. ANGLE 50 C MU .9868



PRESSURE ~ PSIA

$X_{ss}/\text{chord} = \underline{1.19}$ $Y_{ss}/\text{chord} = \underline{.265}$

BETA = 20

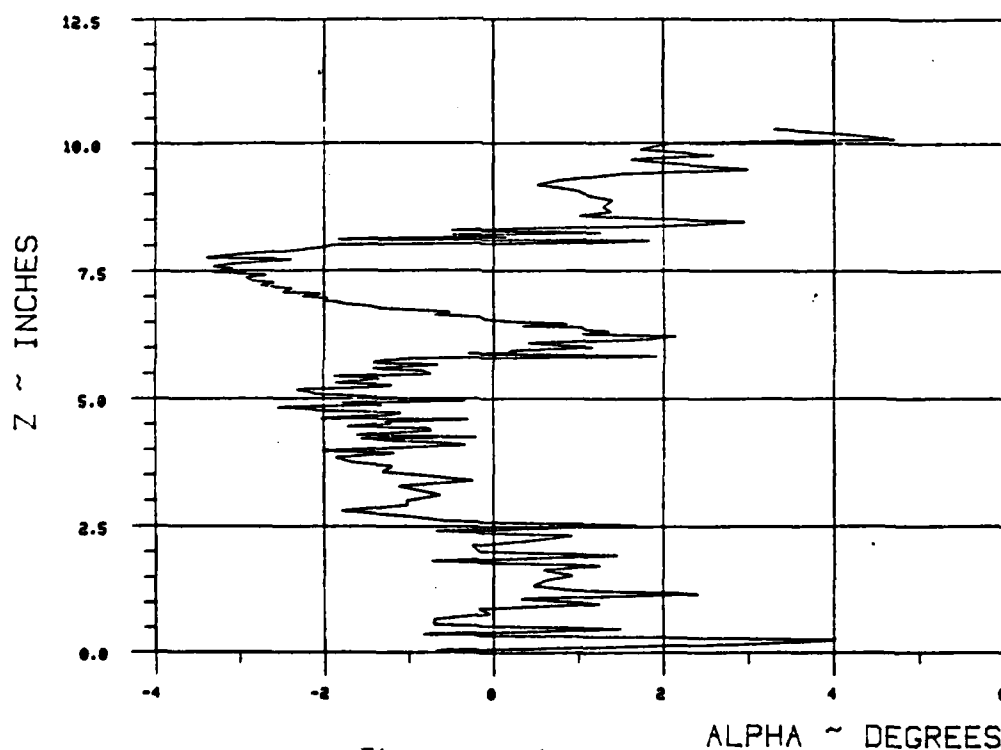
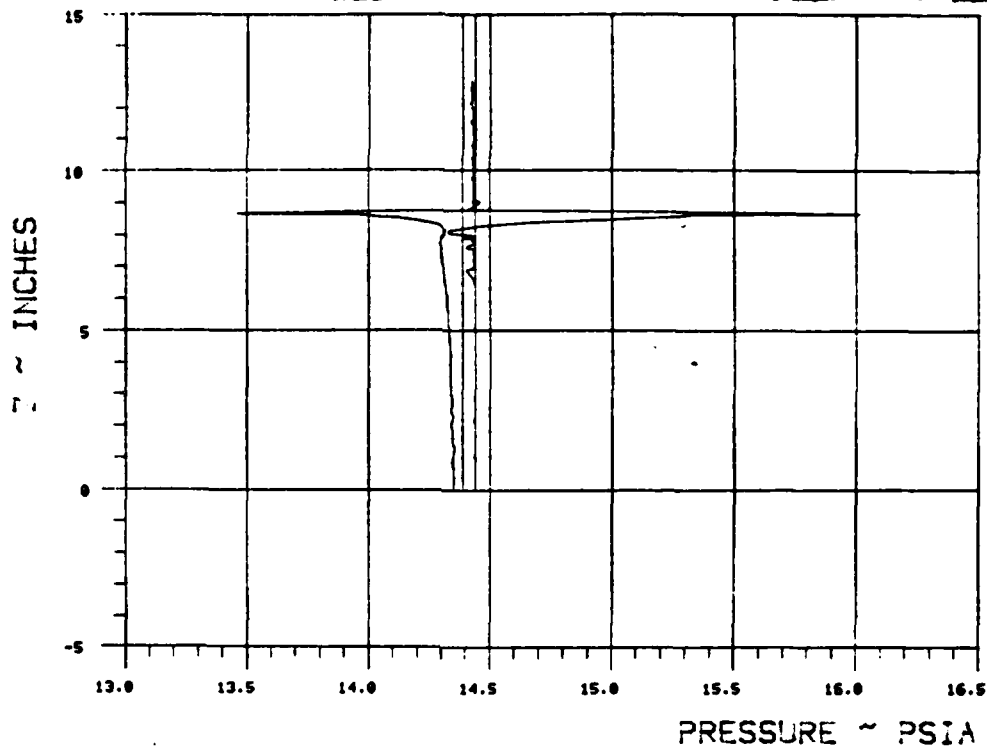


Figure III-36.

WAKE SURVEY DATA

TEST 287 RUN 29

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 1



$X_{ss}/\text{chord} = \underline{.582}$ $Y_{ss}/\text{chord} = \underline{.722}$

BETA = 30

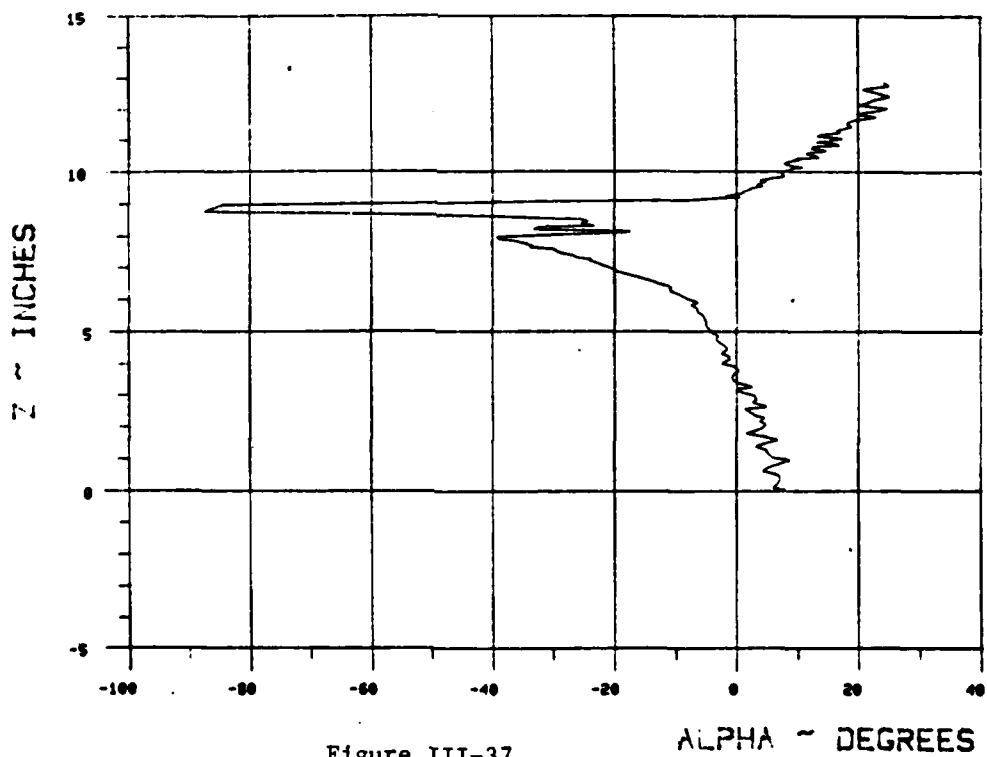
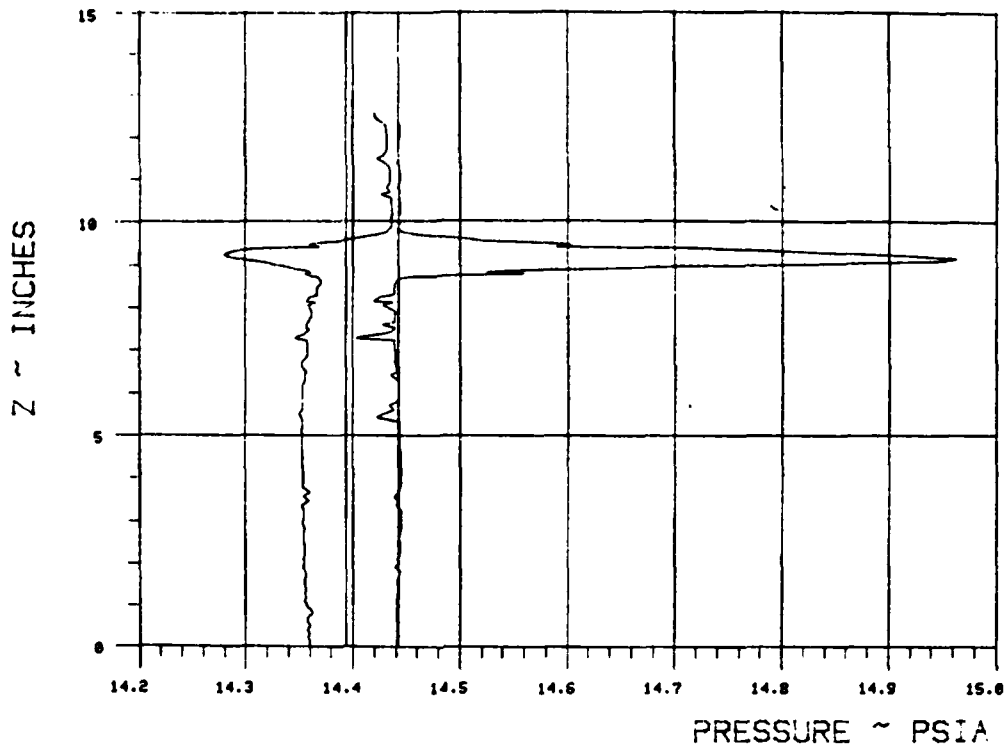


Figure III-37.

WAKE SURVEY DATA

TEST 287 RUN 30

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 1.0512



$X_{se}/\text{chord} = \underline{.69}$ $Y_{se}/\text{chord} = \underline{.66}$

BETA = 30

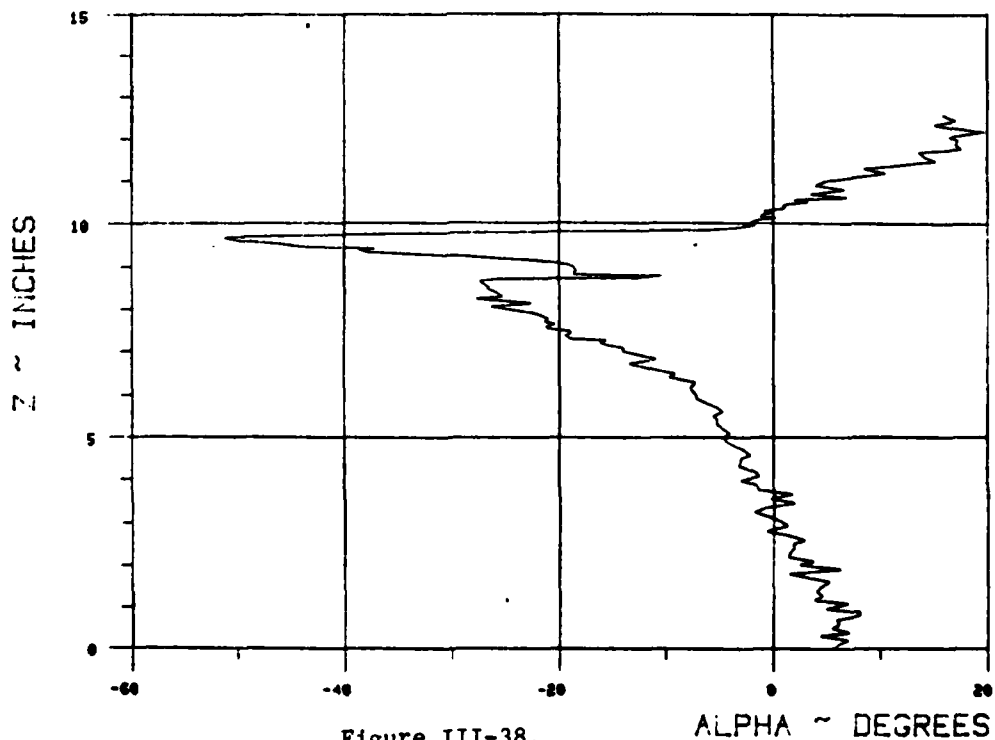
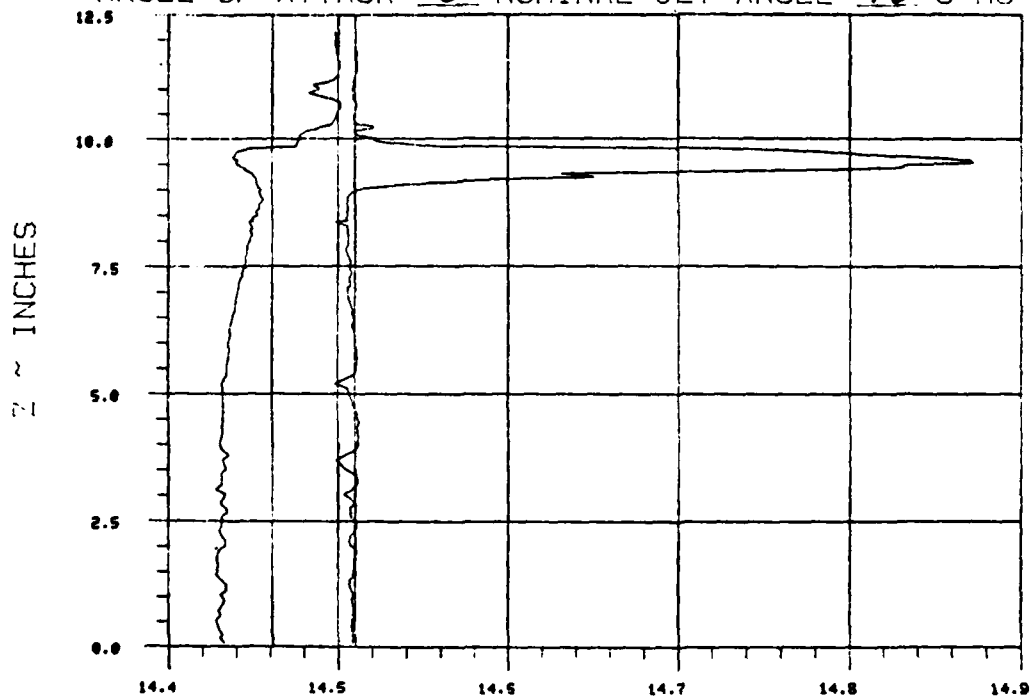


Figure III-38.

WAKE SURVEY DATA

TEST 287 RUN 31

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 1.0314



PRESSURE ~ PSIA

$X_{ss}/\text{chord} = .798$ $Y_{ss}/\text{chord} = .597$

BETA = 30

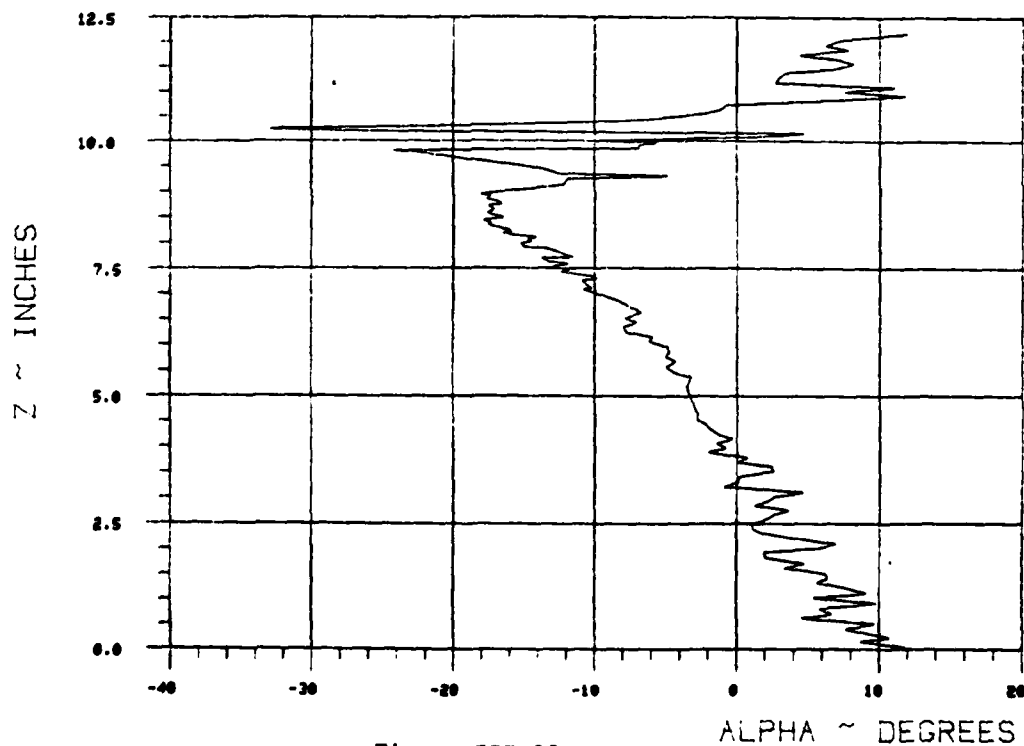
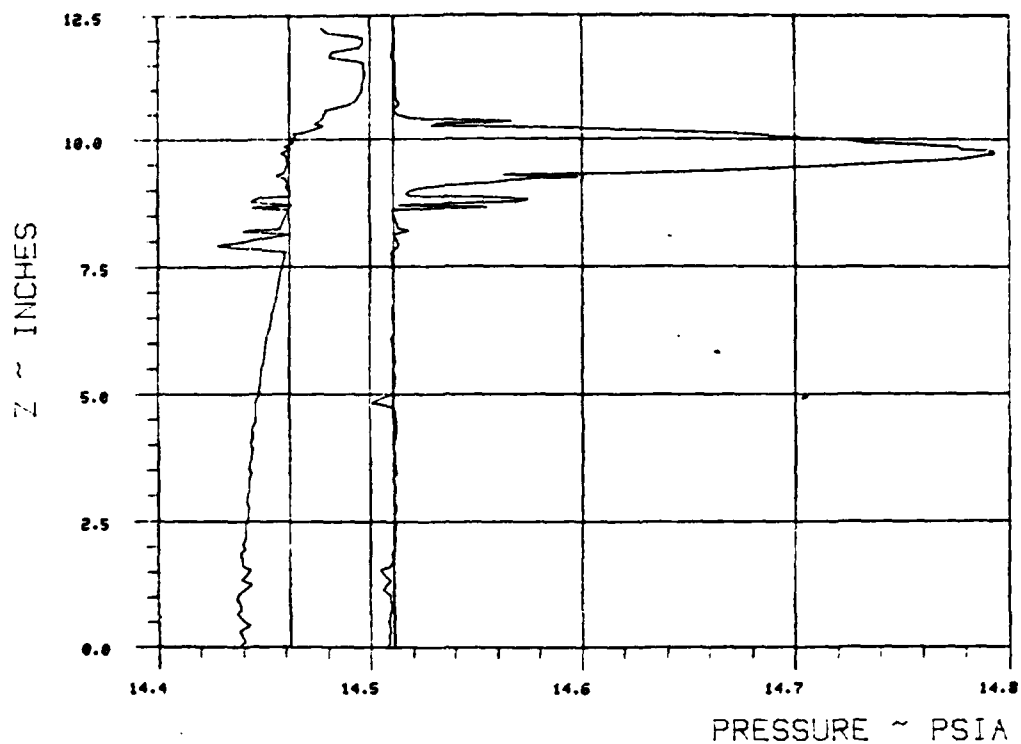


Figure III-39.

WAKE SURVEY DATA

TEST 287 RUN 32

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 1.0394



$X_{ss}/\text{chord} = \underline{1.015}$ $Y_{ss}/\text{chord} = \underline{.472}$

BETA = 30

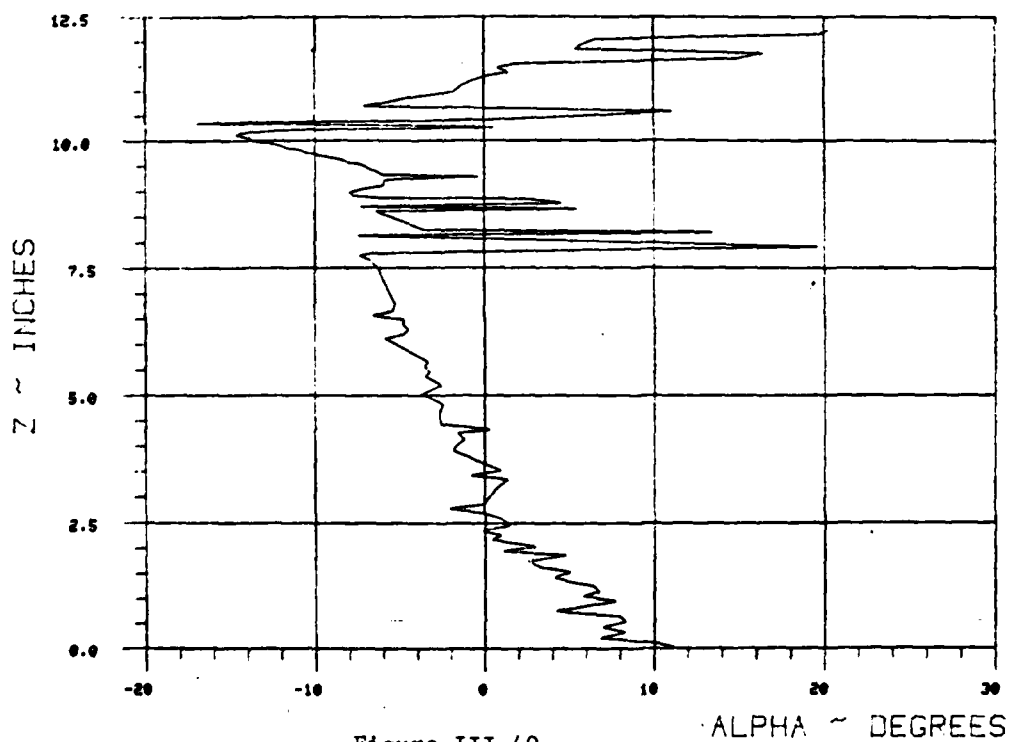
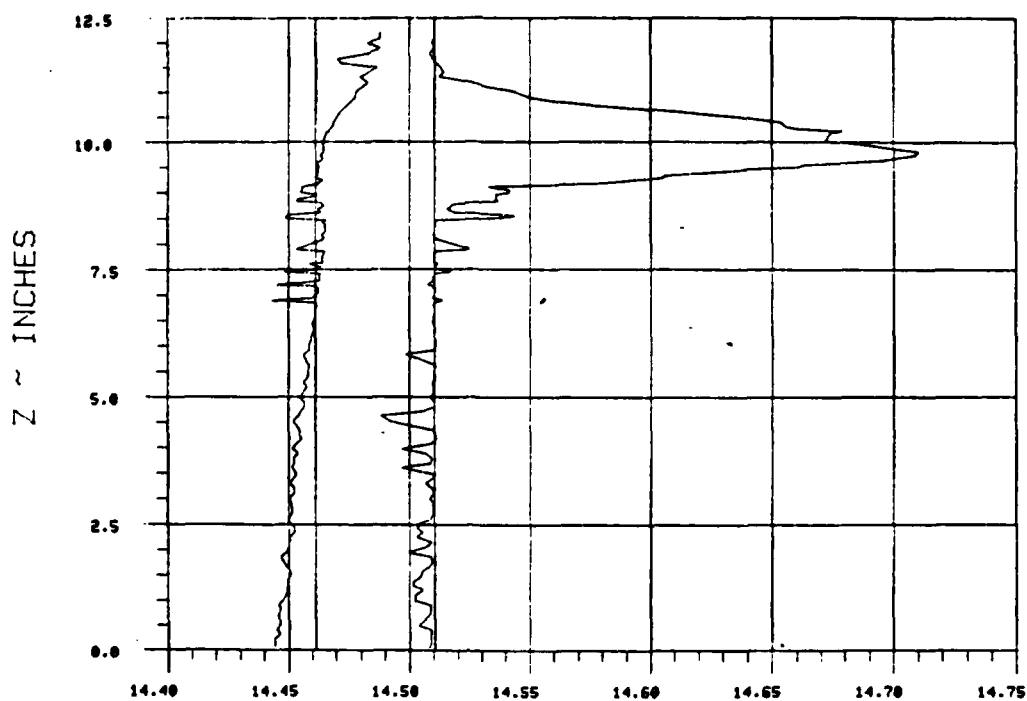


Figure III-40.

WAKE SURVEY DATA

TEST 287 RUN 33

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 1.0639



PRESSURE ~ PSIA

$X_{ss}/\text{chord} = \underline{1.34}$ $Y_{ss}/\text{chord} = \underline{.285}$

BETA = 30

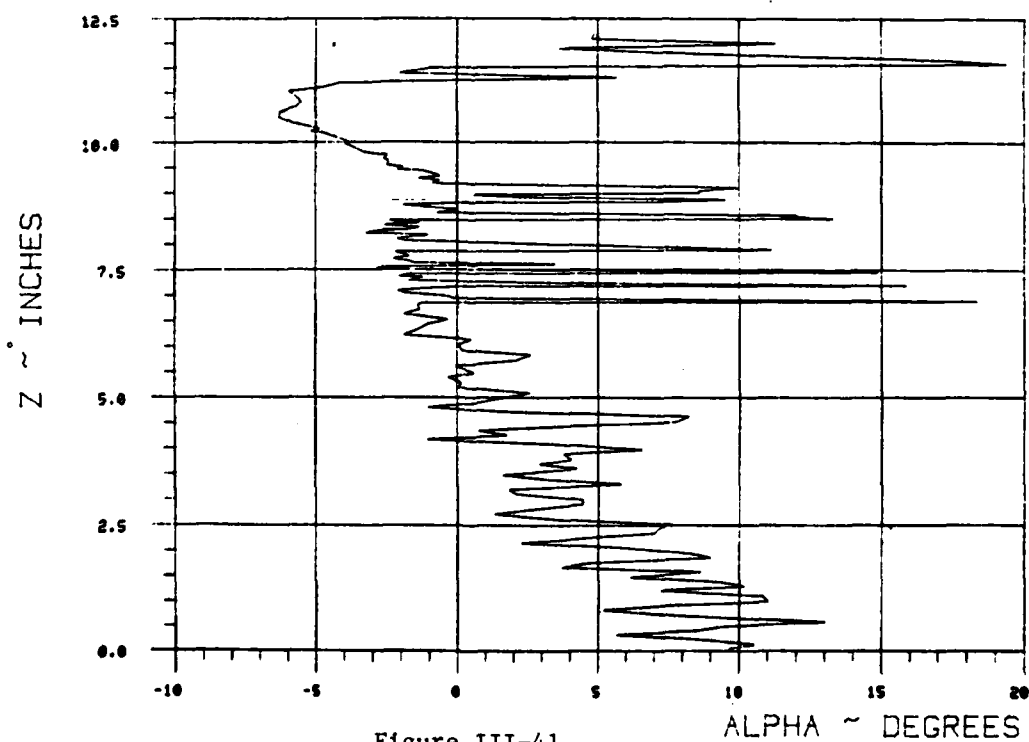
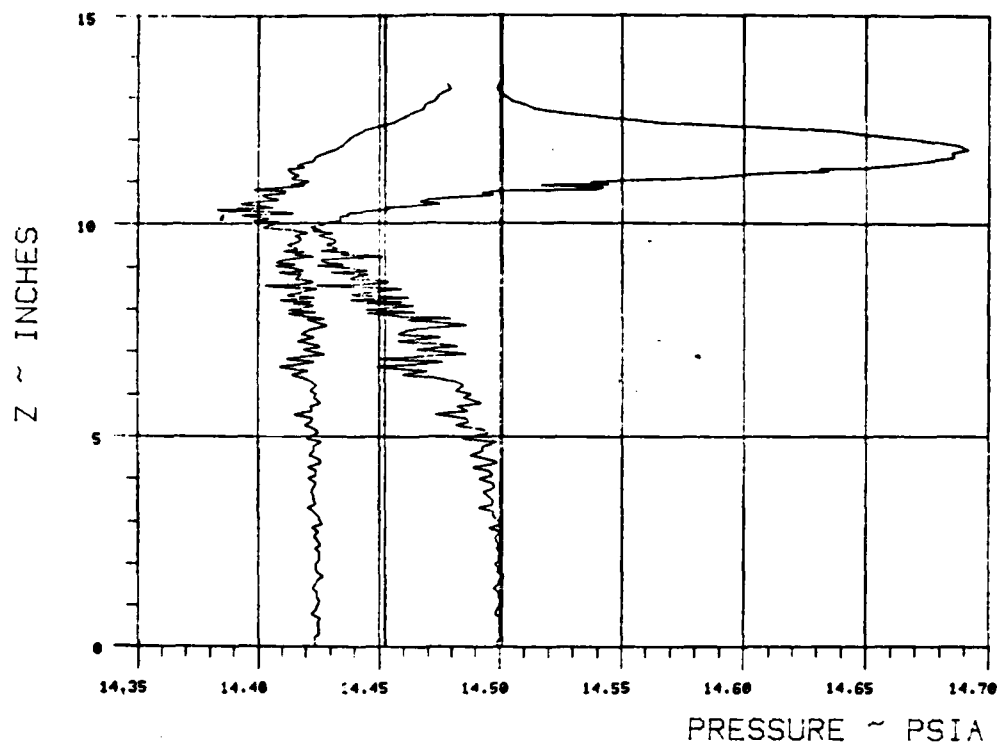


Figure III-41.

WAKE SURVEY DATA

TEST 287 RUN 34

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 2.1181



$X_{ss}/\text{chord} = \underline{1.14}$ $Y_{ss}/\text{chord} = \underline{.688}$

BETA = 30

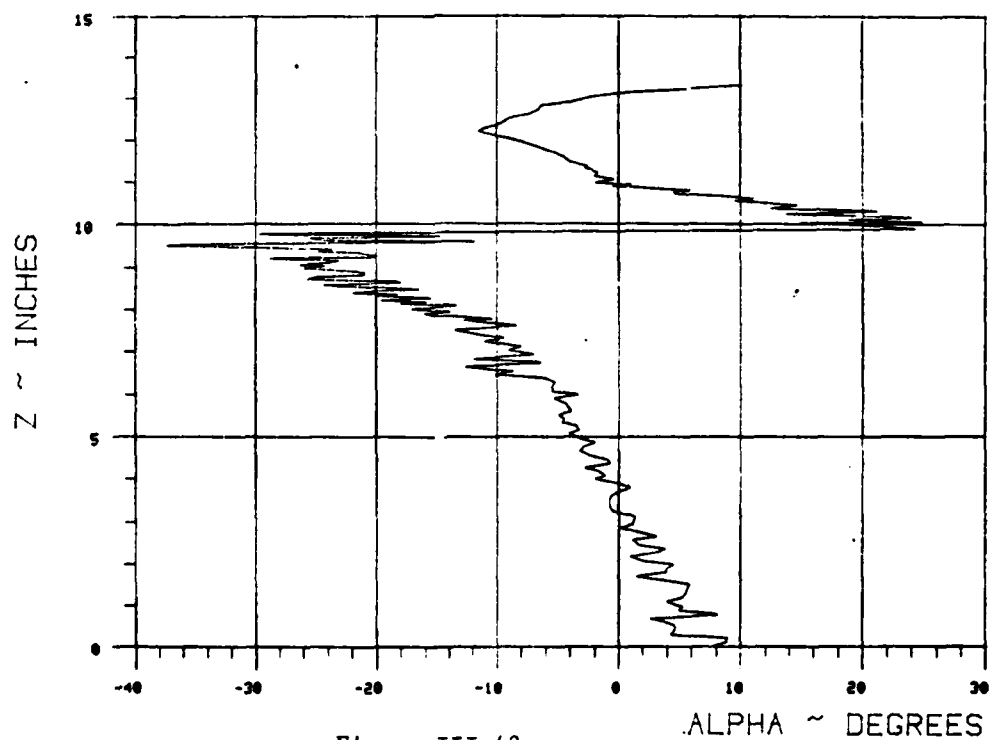
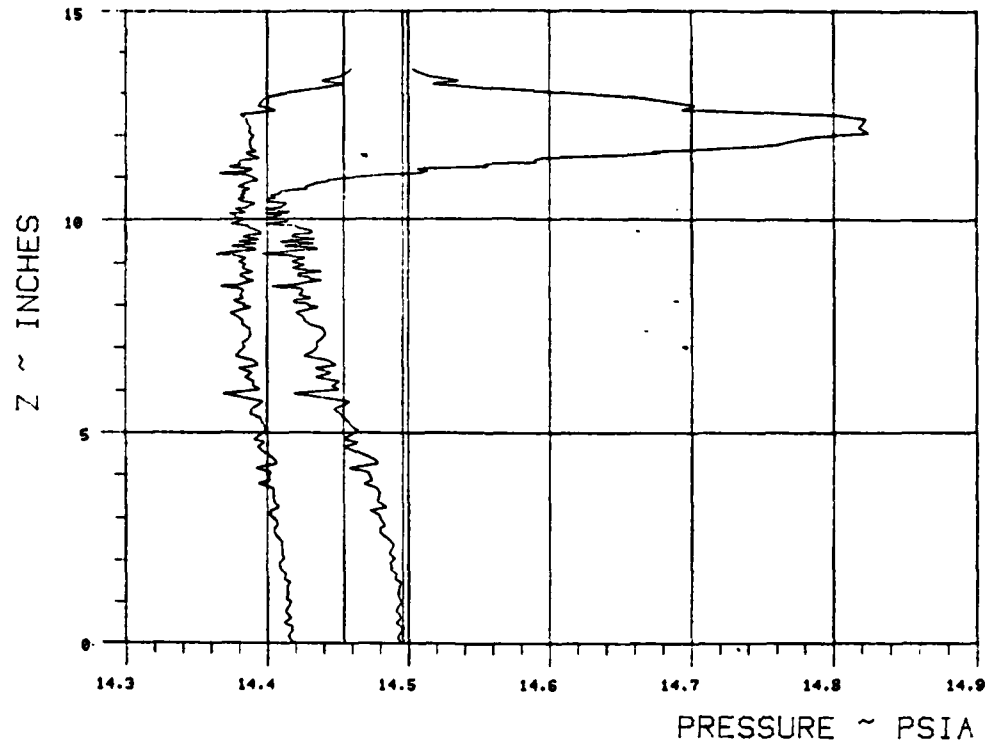


Figure III-42.

WAKE SURVEY DATA

TEST 207 RUN 35

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 70 C MU 4.1694



$X_{ss}/\text{chord} = \underline{1.14}$ $Y_{ss}/\text{chord} = \underline{.688}$

BETA = 30

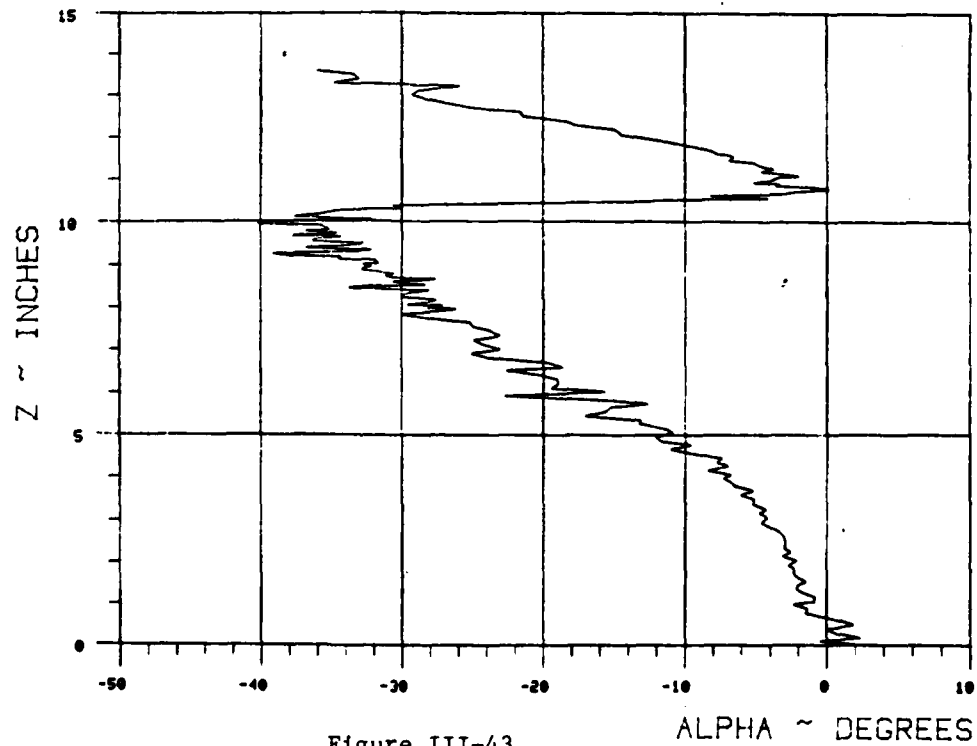
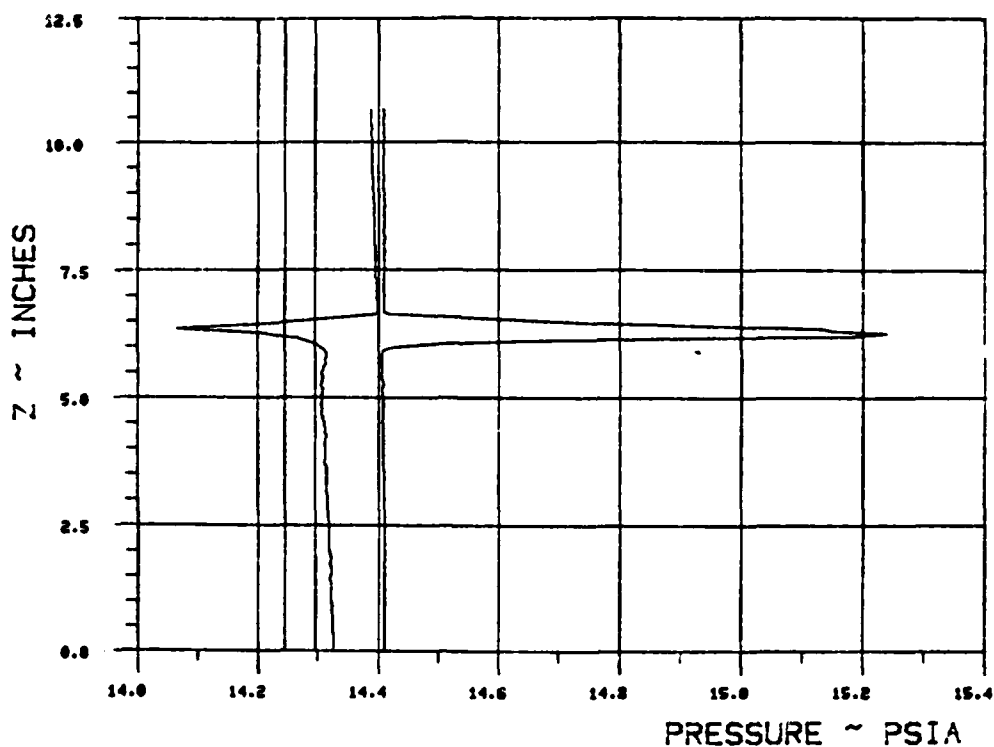


Figure III-43.

WAKE SURVEY DATA

TEST 287 RUN 1

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU .9412



$X_{ss}/\text{chord} = .6733$ $Y_{ss}/\text{chord} = .7222$

BETA = 30

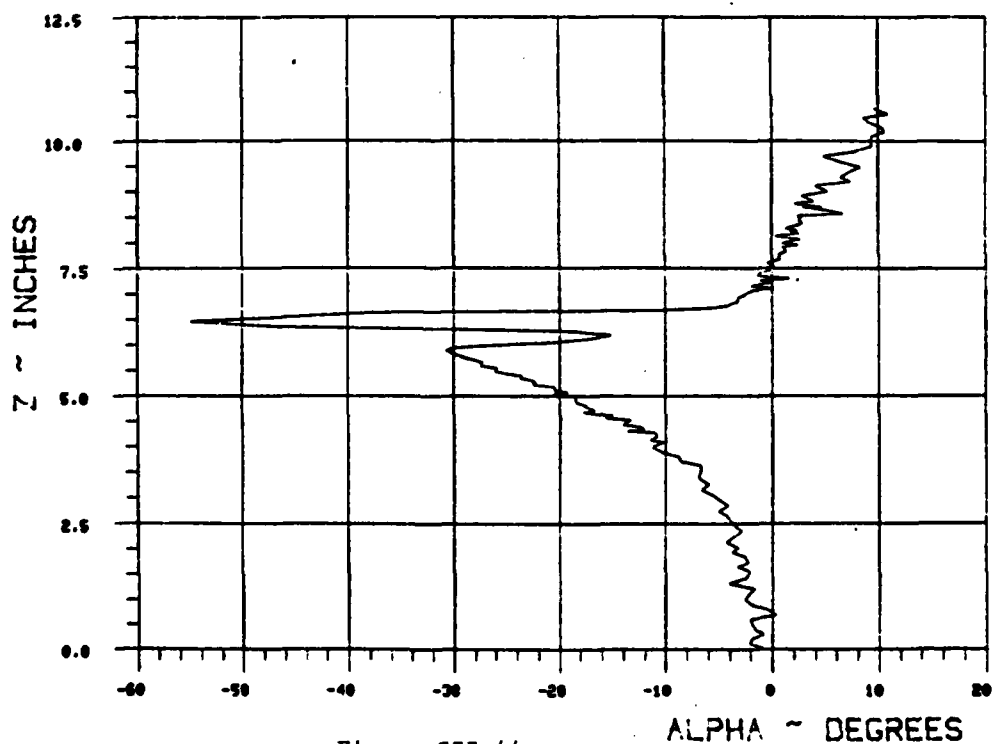
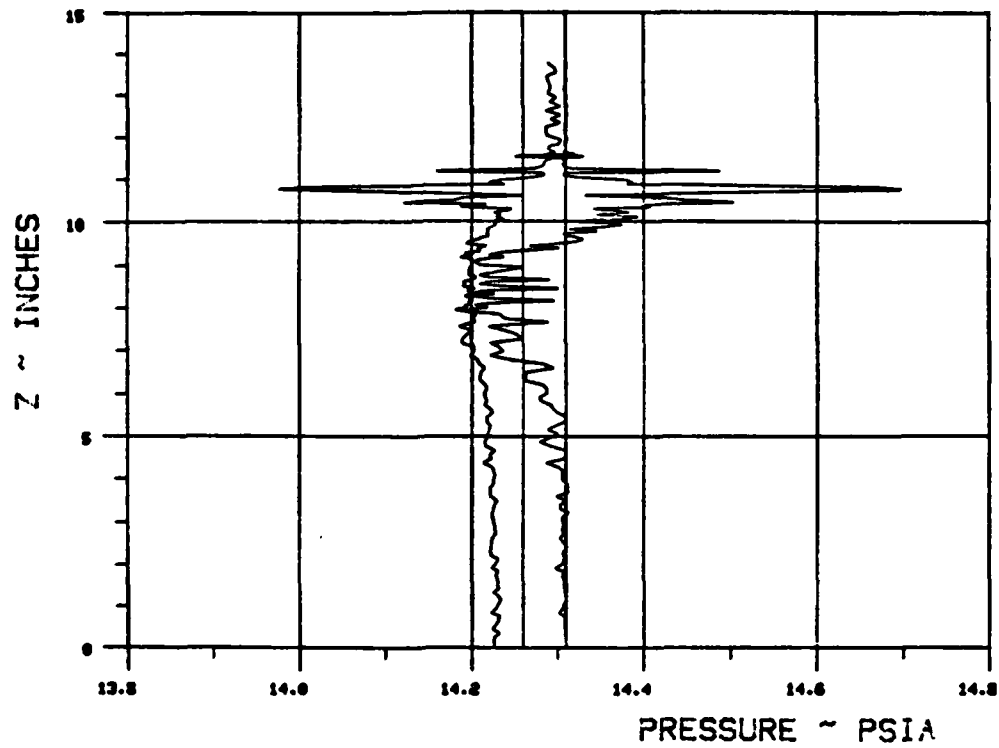


Figure III-44.

WAKE SURVEY DATA

TEST 287 RUN 2

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU .9367



$X_{ss}/\text{chord} = .7816$ $Y_{ss}/\text{chord} = .6597$

BETA = 30

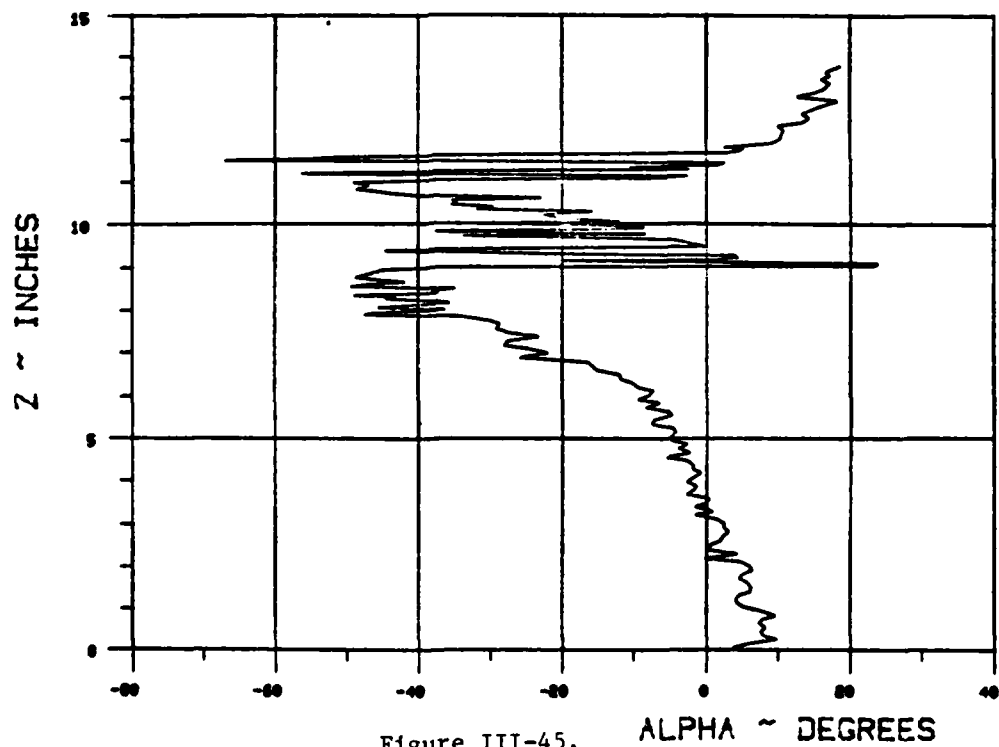
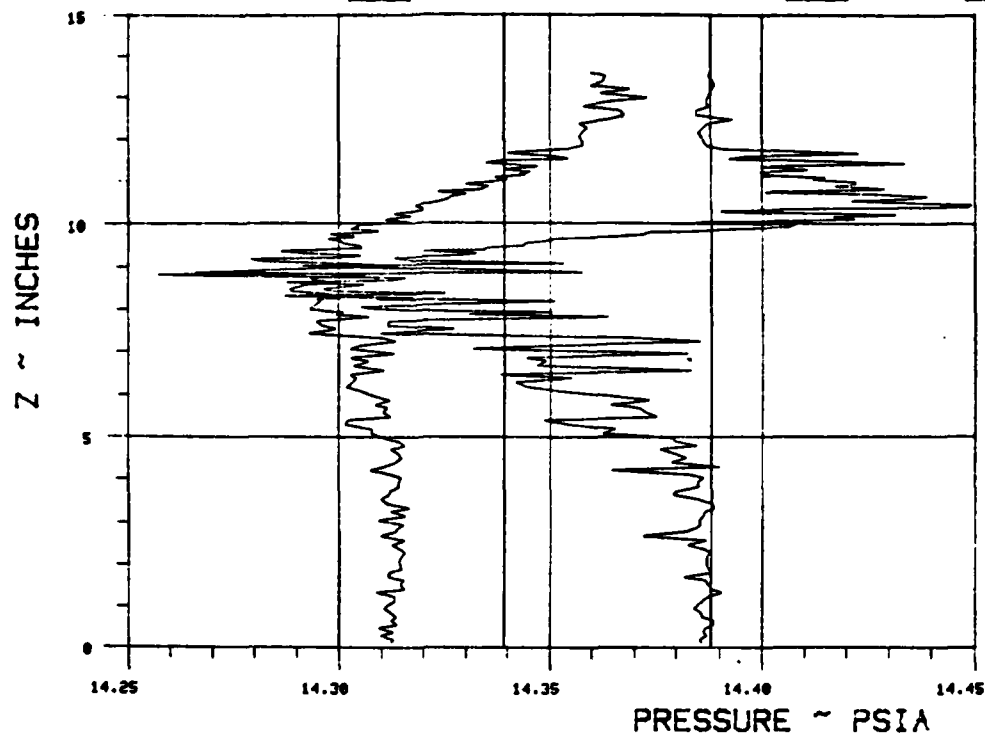


Figure III-45.

WAKE SURVEY DATA

TEST 287 RUN 3

ANGLE OF ATTACK 6 NOMINAL JET ANGLE 90 C MU .9319



$X_{ee}/\text{chord} = .8898$ $Y_{ee}/\text{chord} = .5972$

BETA = 30

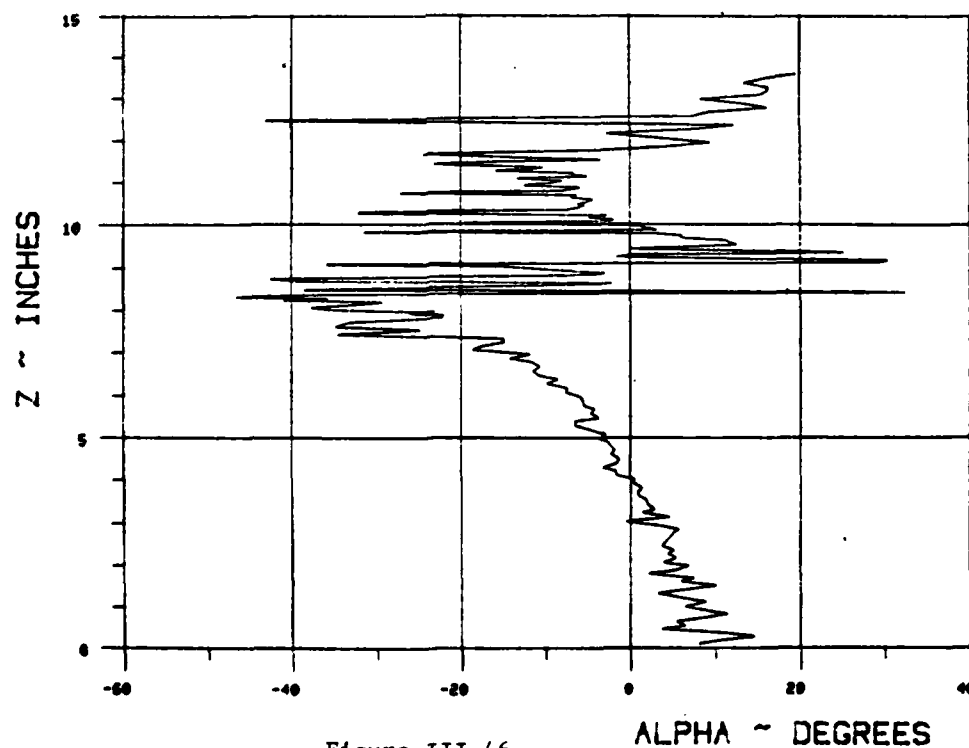
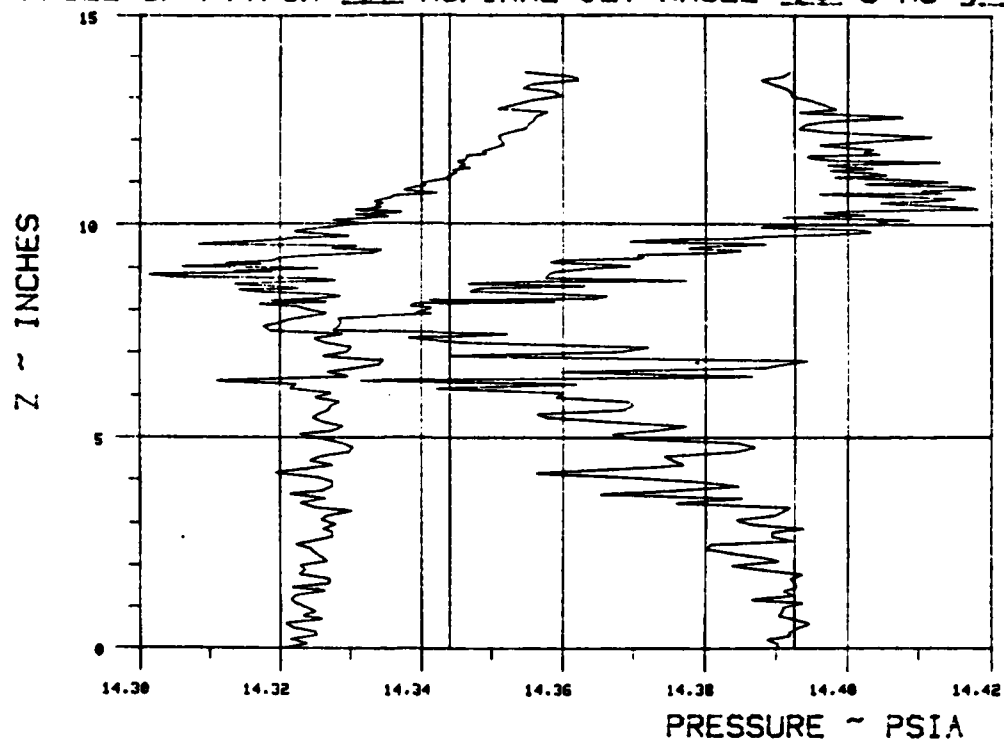


Figure III-46.

WAKE SURVEY DATA

TEST 287 RUN 4

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU .9774



$X_{ss}/\text{chord} = \underline{1.1064}$ $Y_{ss}/\text{chord} = \underline{.4722}$

BETA = 30

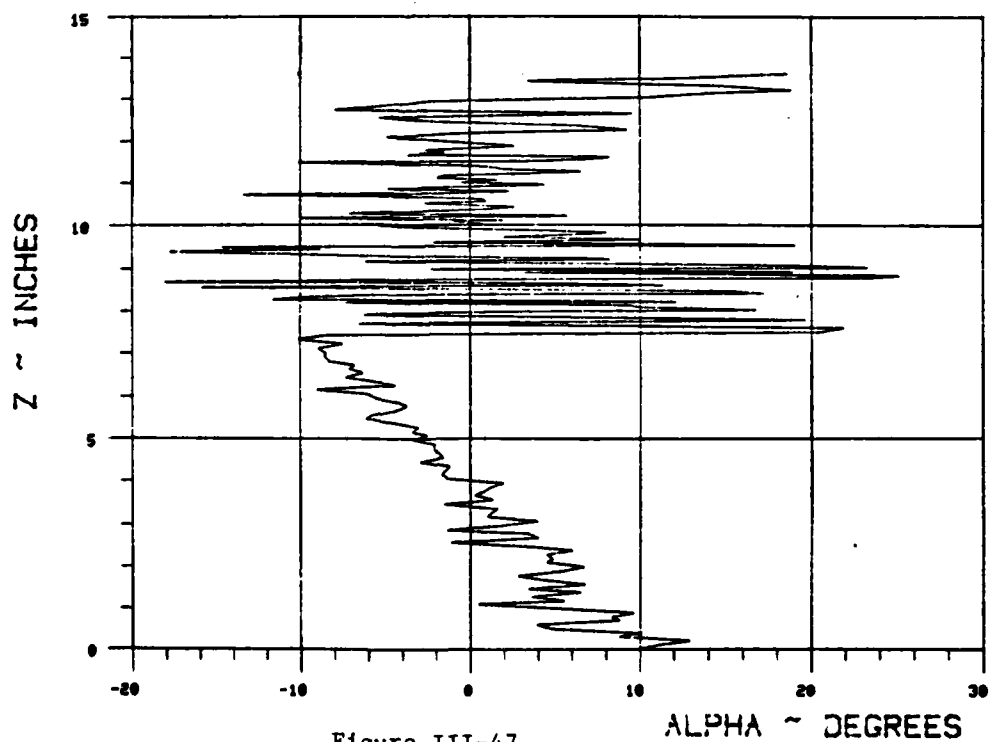
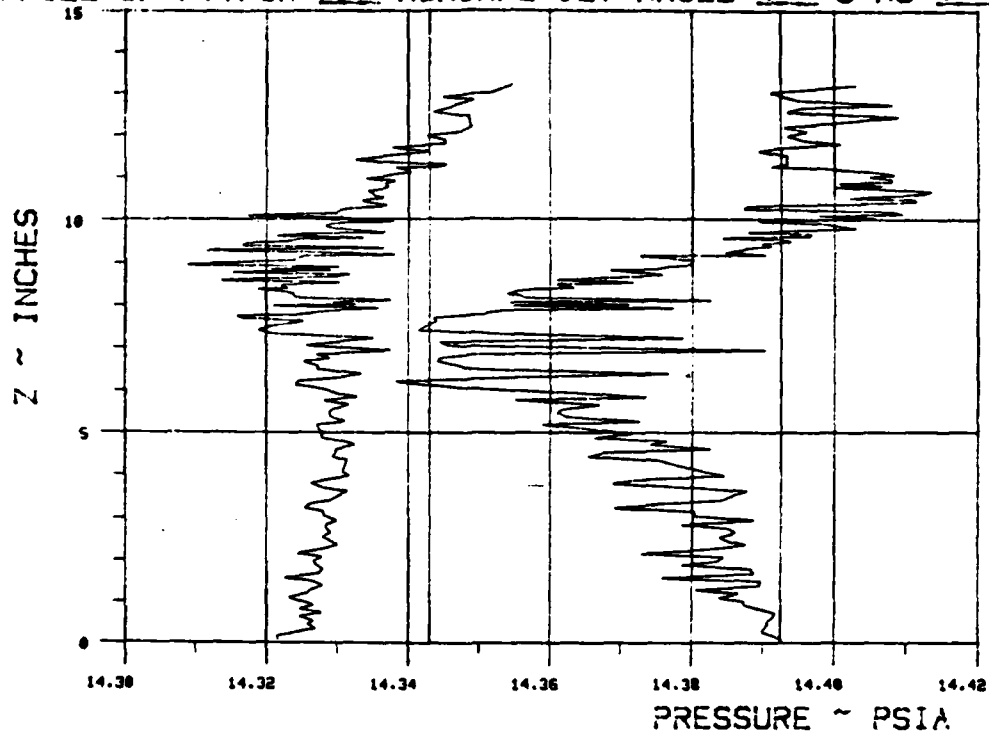


Figure III-47.

WAKE SURVEY DATA

TEST 287 RUN 5

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU .9256



$X_{ee}/\text{chord} = \underline{.1246}$ $Y_{ee}/\text{chord} = \underline{.4097}$

BETA = 30

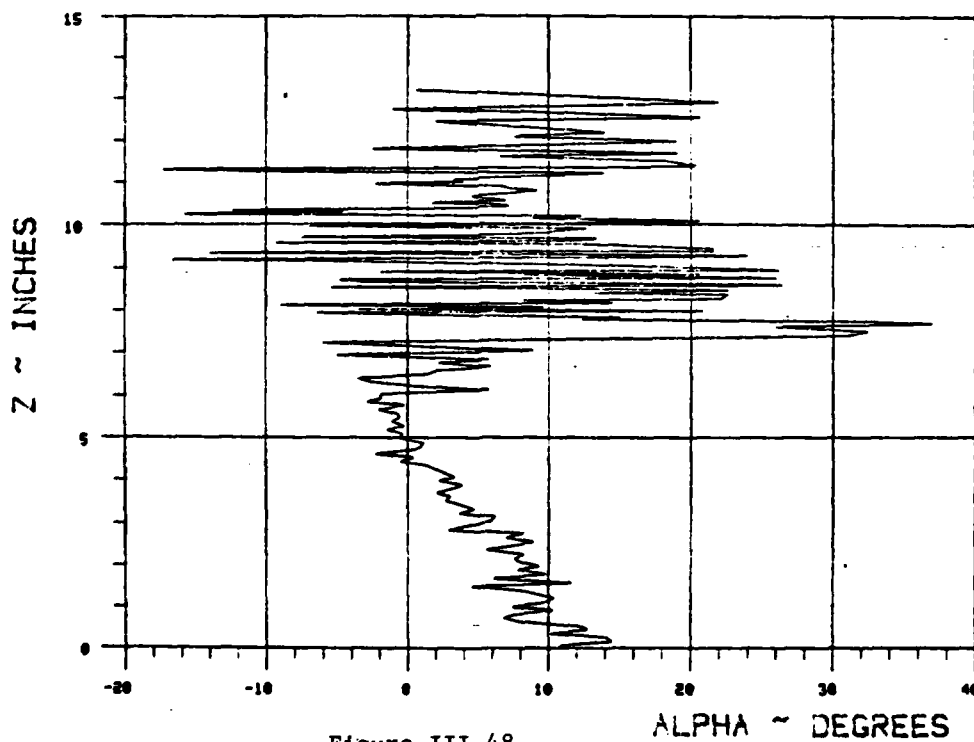
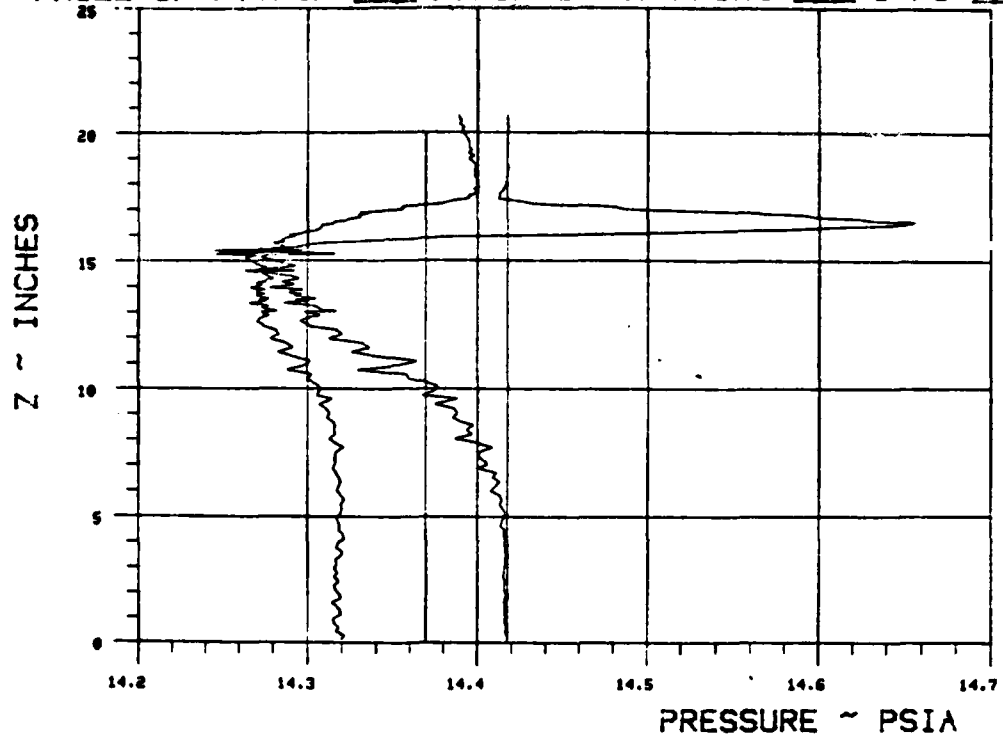


Figure III-48.

WAKE SURVEY DATA

TEST 287 RUN 6

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 1.88/6



$X_{ee}/\text{chord} = \underline{1.4419}$ $Y_{ee}/\text{chord} = \underline{1.1244}$

BETA = 45

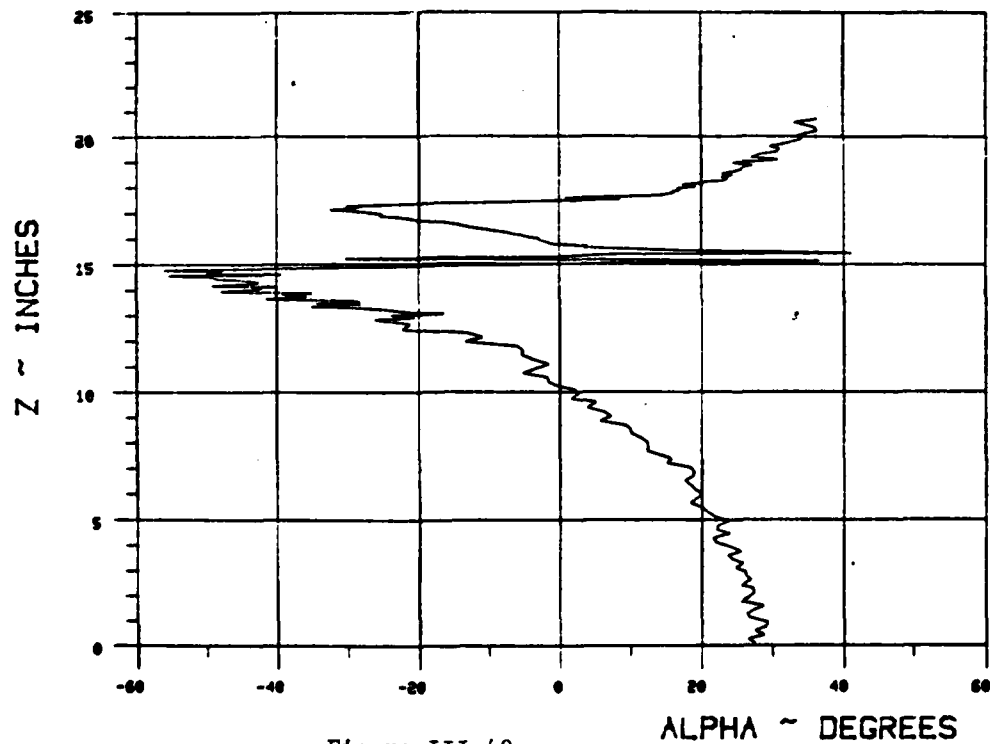
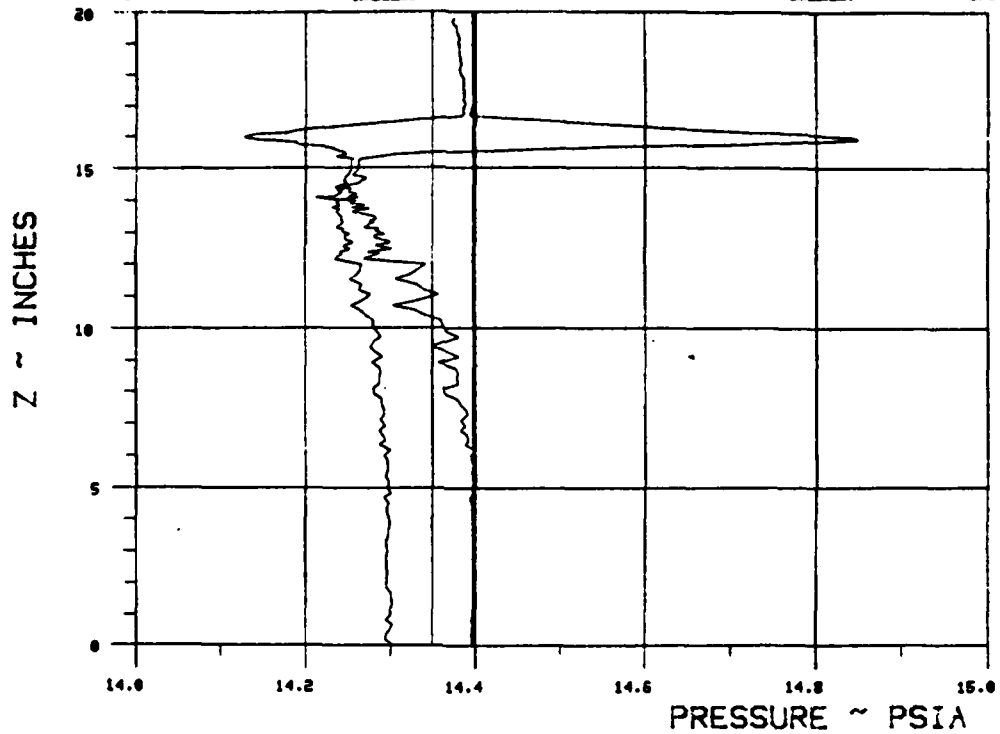


Figure III-49.

WAKE SURVEY DATA

TEST 287 RUN I

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 1.9165



$X_{ss}/\text{chord} = \underline{1.5303}$ $Y_{ss}/\text{chord} = \underline{1.036}$

BETA = 45

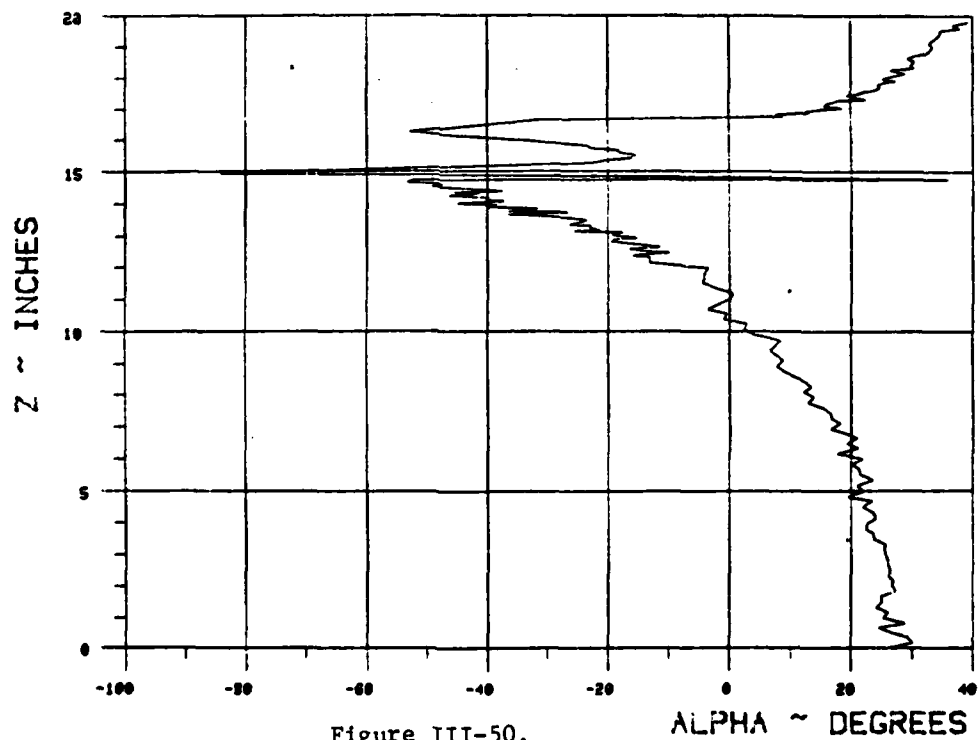
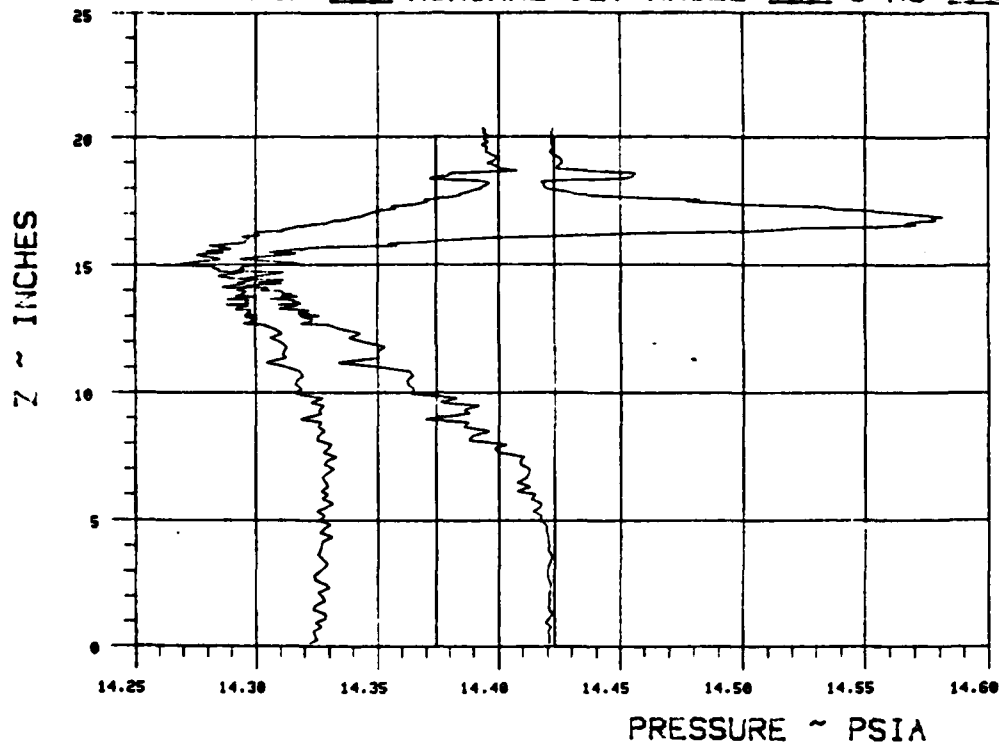


Figure III-50.

WAKE SURVEY DATA

TEST 287 RUN 8

ANGLE OF ATTACK 6 NOMINAL JET ANGLE 90 C MU 1.8849



$X_{ss}/\text{chord} = \underline{1.6187}$ $Y_{ss}/\text{chord} = \underline{.9477}$

BETA = 45

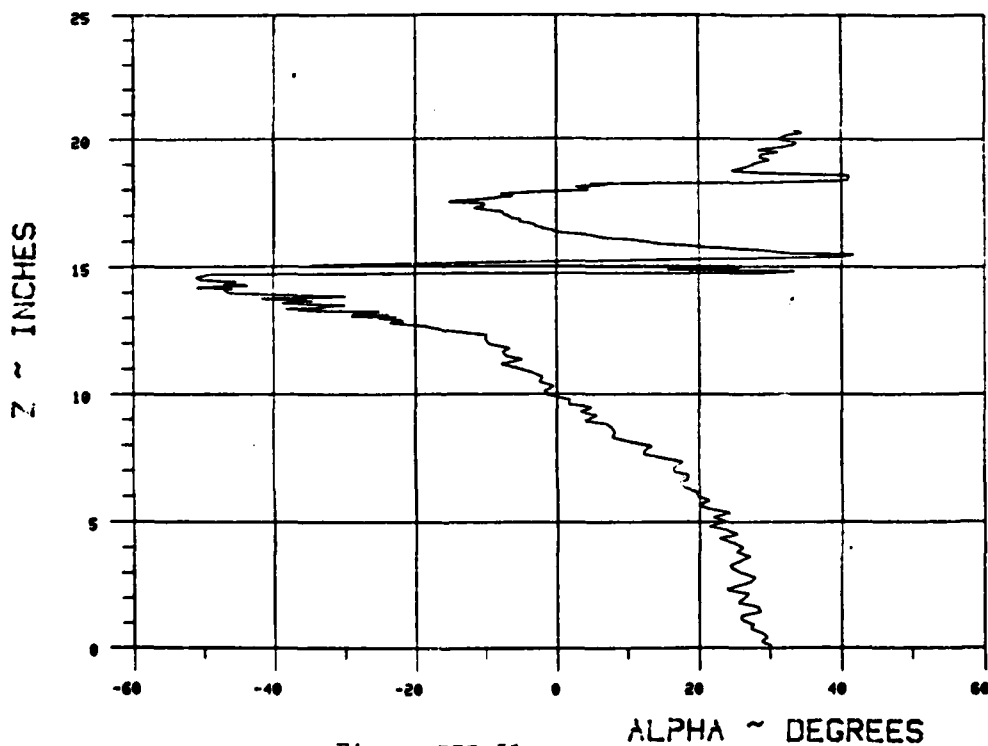
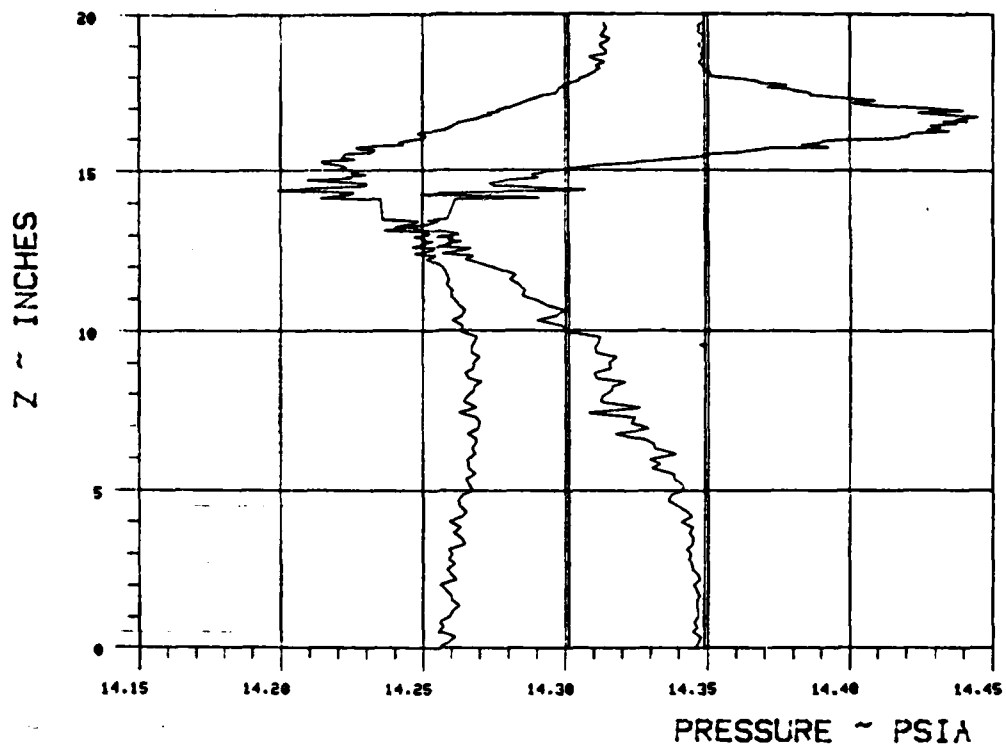


Figure III-51.

WAKE SURVEY DATA

TEST 287 RUN 9

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 1.9428



$X_{ee}/\text{chord} = 1.7955$ $Y_{ee}/\text{chord} = .7709$

BETA = 45

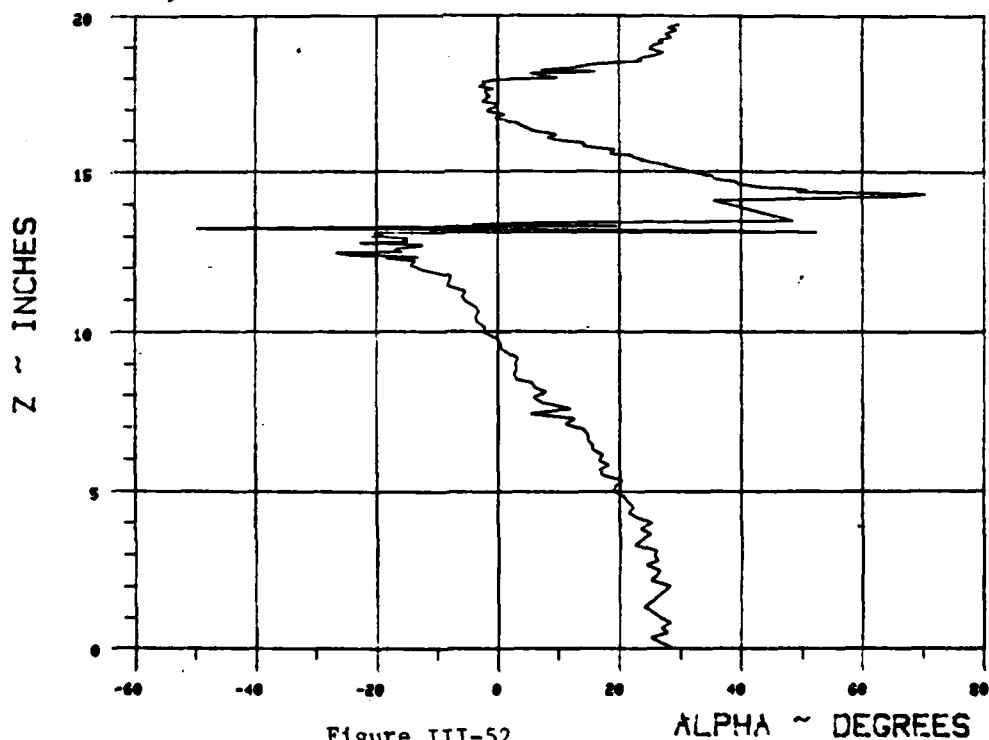
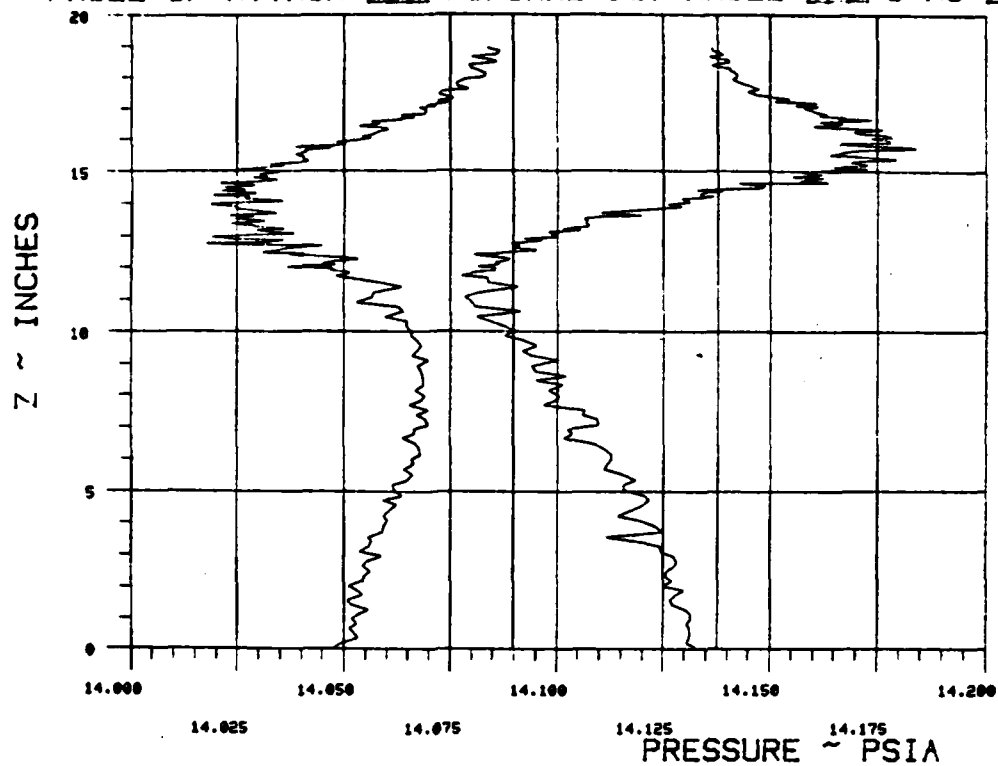


Figure III-52.

WAKE SURVEY DATA

TEST 287 RUN 10

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 1.8989



$X_{ss}/\text{chord} = \underline{2.0607}$ $Y_{ss}/\text{chord} = \underline{.5057}$

BETA = 45

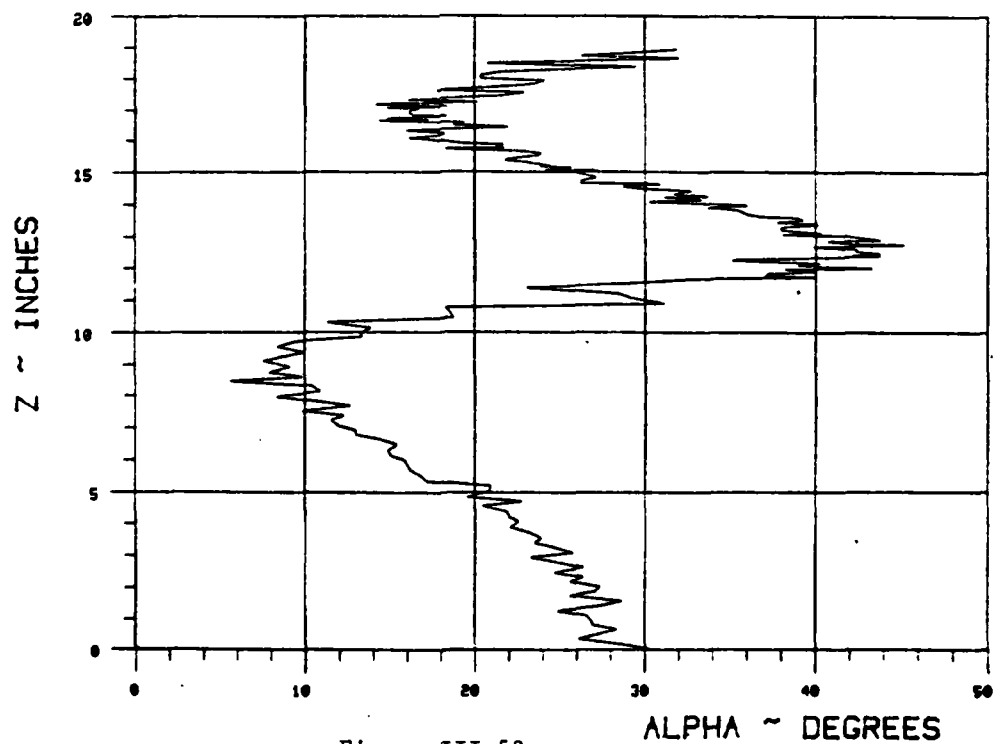
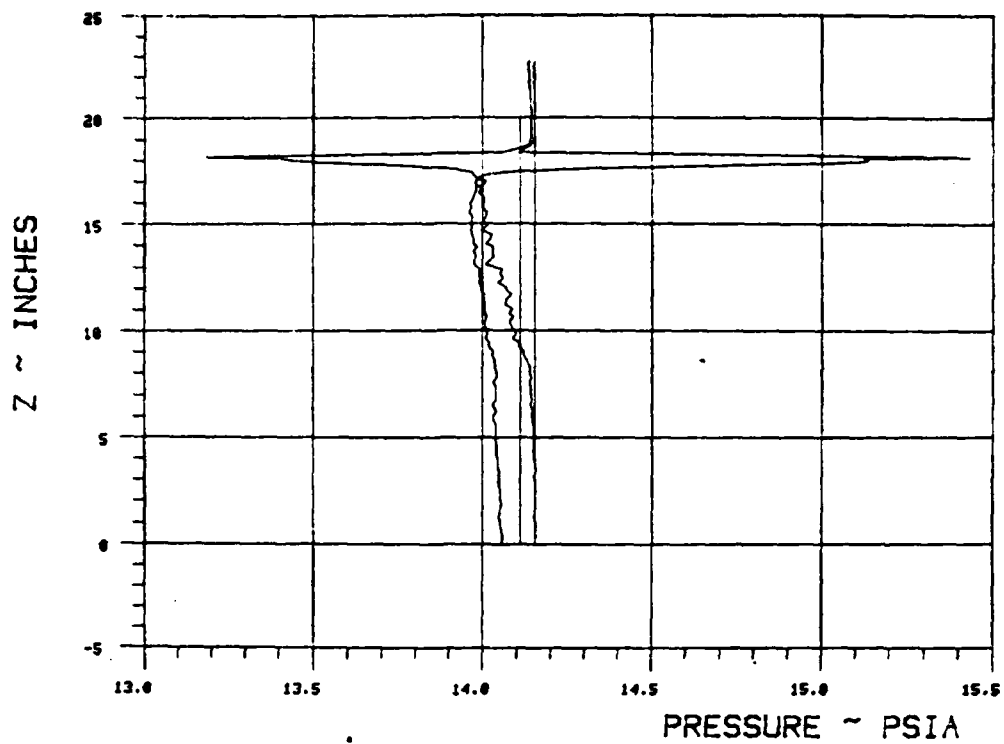


Figure III-53.

WAKE SURVEY DATA

TEST 287 RUN 11

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 3.7314



$X_{ss}/\text{chord} = \underline{1.6189}$ $Y_{ss}/\text{chord} = \underline{1.3014}$

BETA = 45

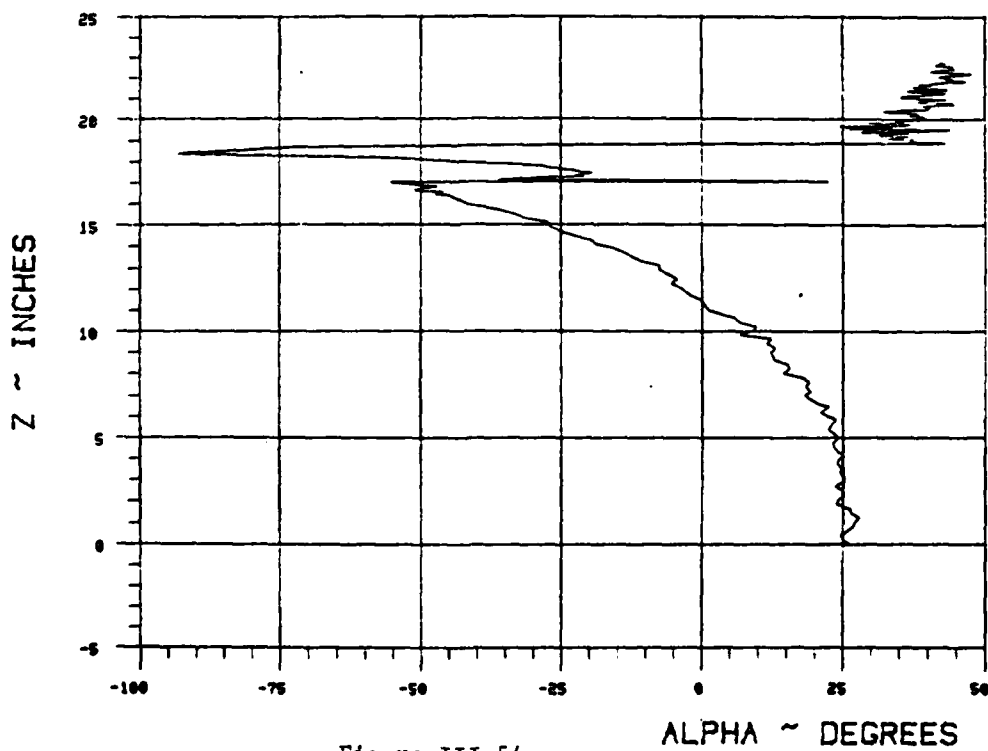
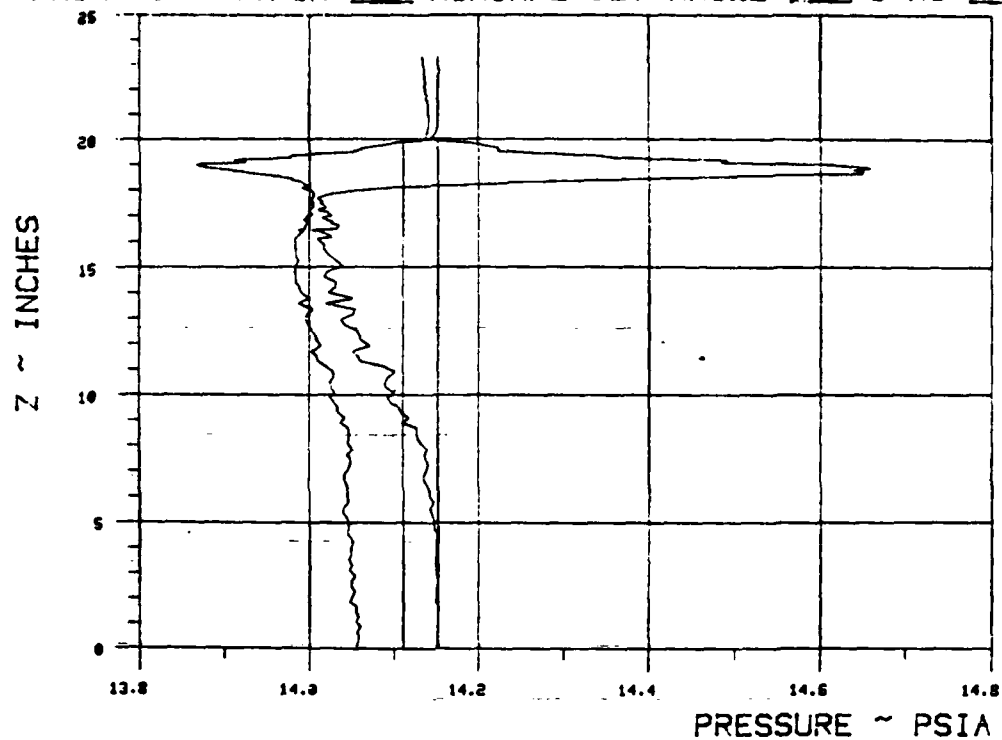


Figure III-54.

WAKE SURVEY DATA

TEST 287 RUN 12

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 3.7542



$X_{ss}/\text{chord} = 1.7073$ $Y_{ss}/\text{chord} = 1.213$

BETA = 45

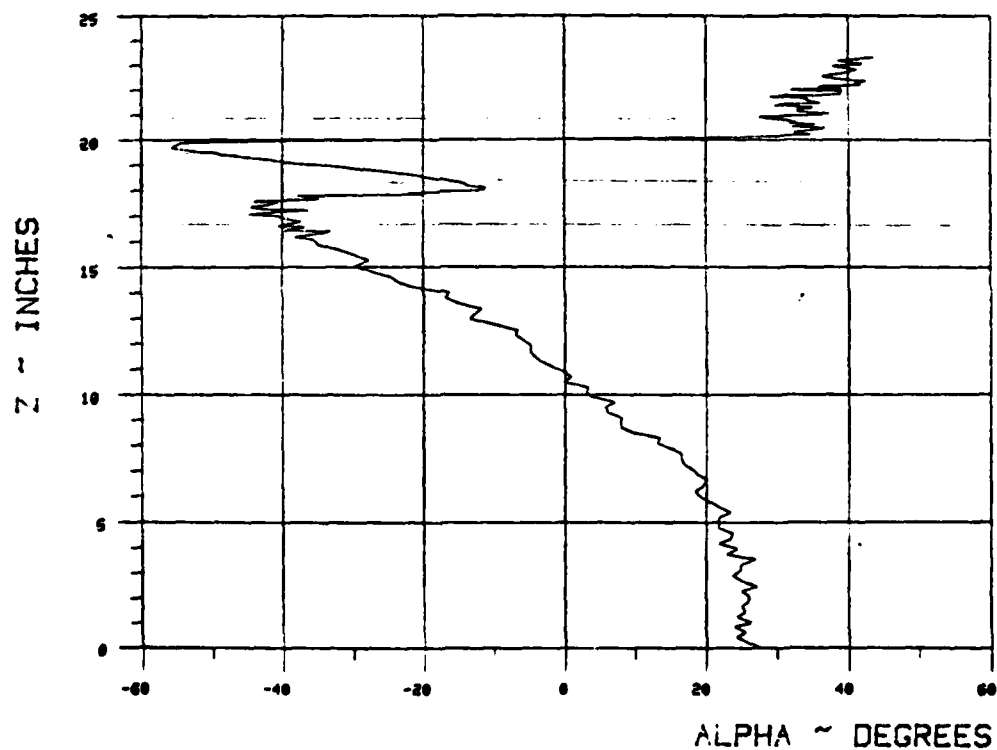
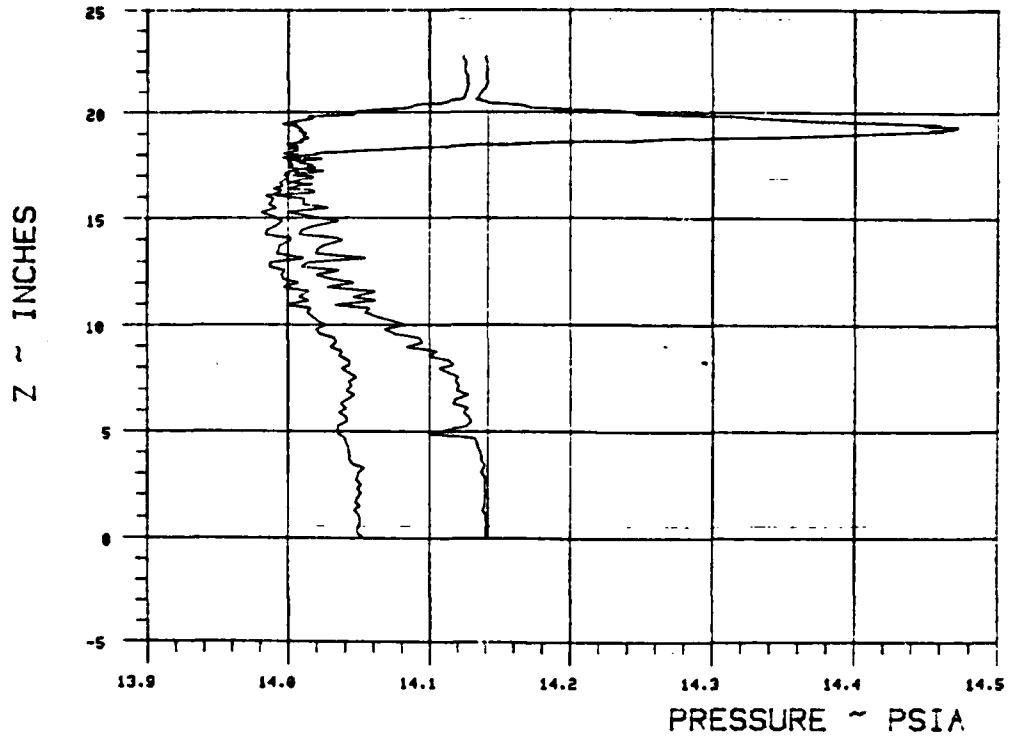


Figure III-55.

WAKE SURVEY DATA

TEST 287 RUN 13

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 3.741



$X_{ss}/\text{chord} = 1.7956$ $Y_{ss}/\text{chord} = 1.1246$

BETA = 45

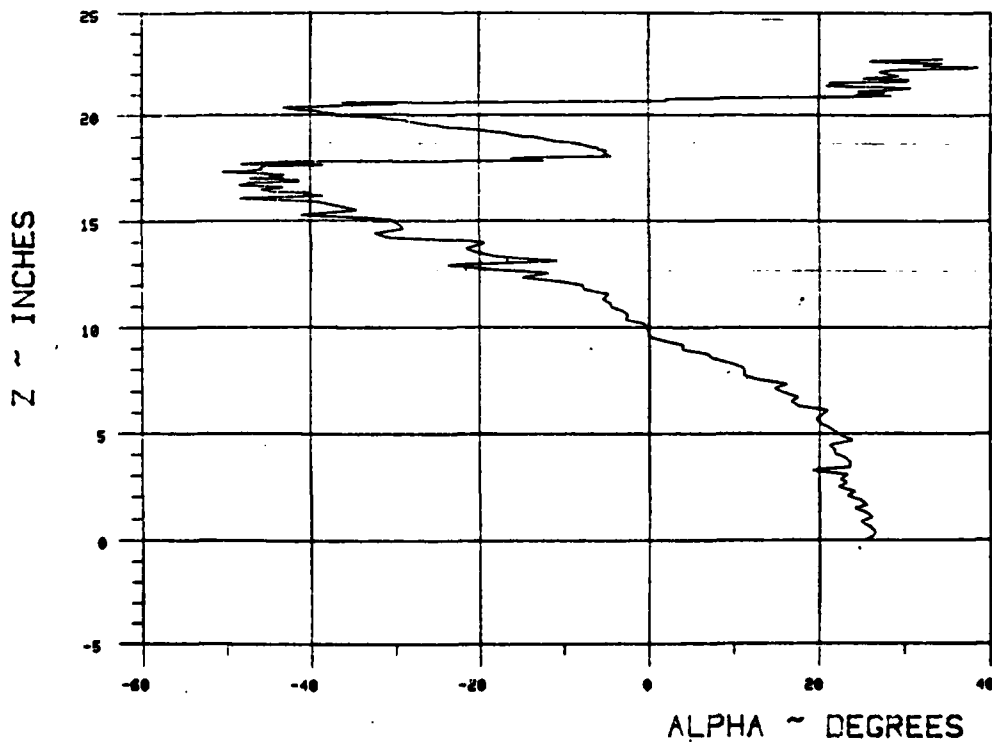
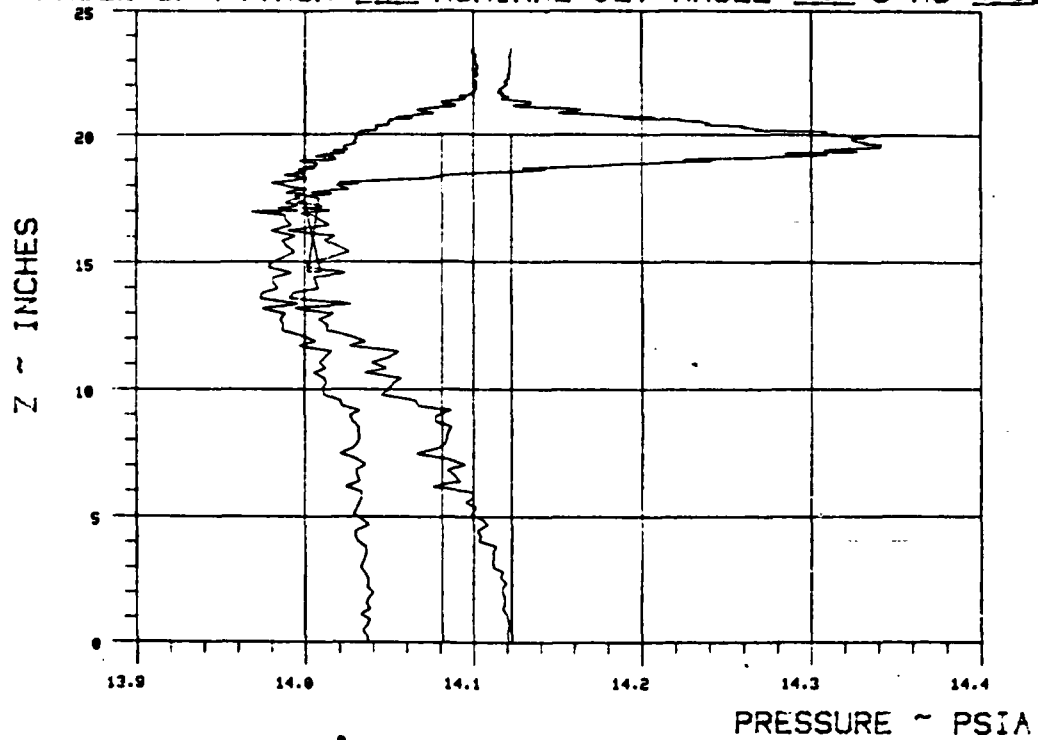


Figure III-56.

WAKE SURVEY DATA

TEST 287 RUN 14

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 3.7669



$X_{ss}/\text{chord} = 1.9724$ $Y_{ss}/\text{chord} = .9478$

BETA = 45

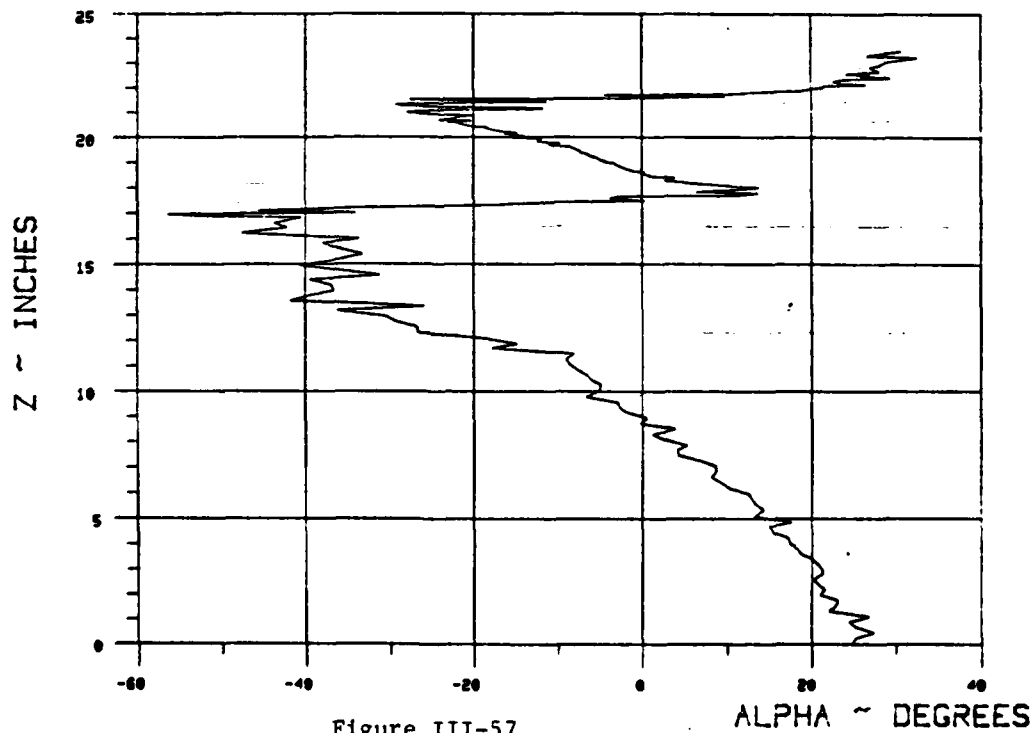
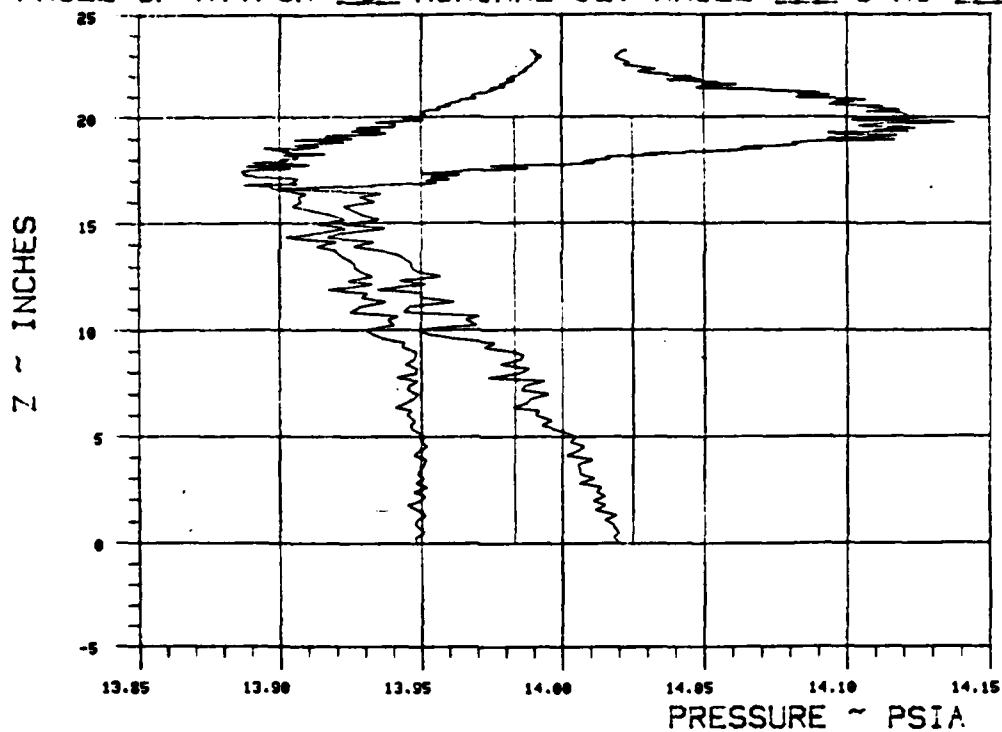


Figure III-57.

WAKE SURVEY DATA

TEST 287 RUN 15

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 3.6752



$X_{ee}/\text{chord} = 2.2376$ $Y_{ee}/\text{chord} = .6827$

BETA = 45

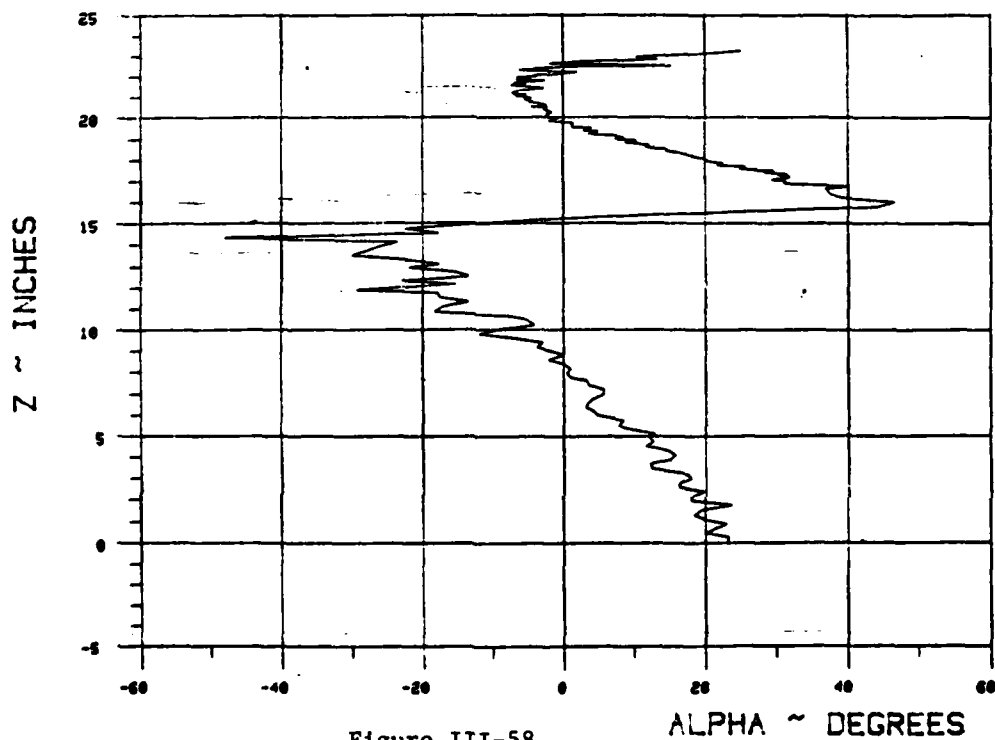
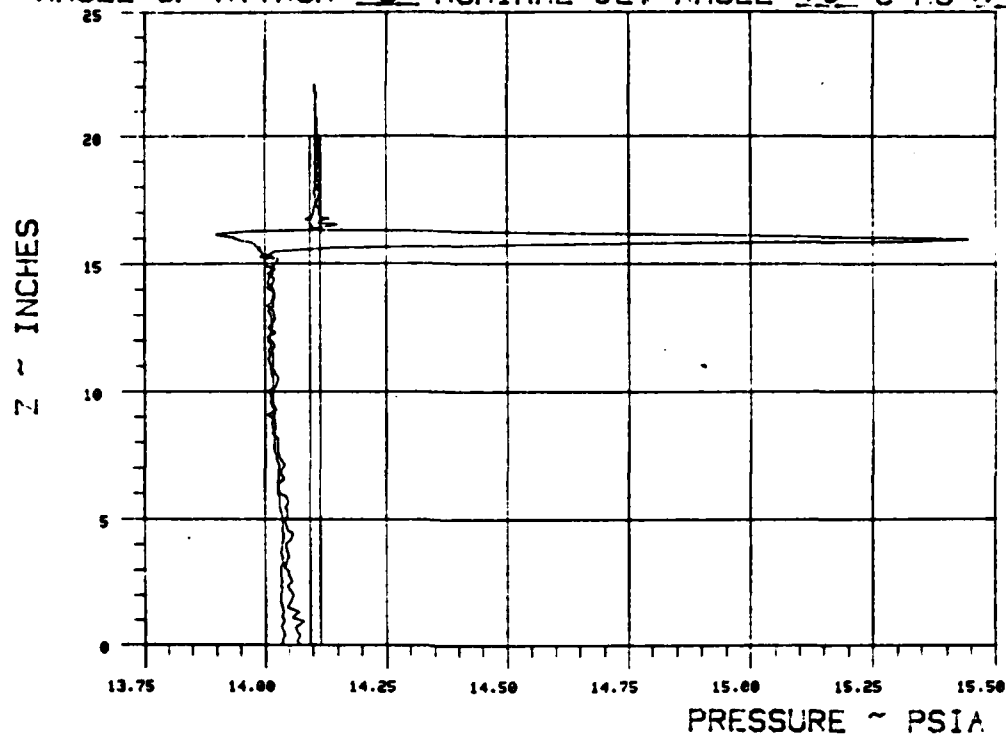


Figure III-58.

WAKE SURVEY DATA

TEST 287 RUN 16

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 7.931



$X_{se}/chord = \underline{1.857}$ $Y_{se}/chord = \underline{.382}$

BETA = 70

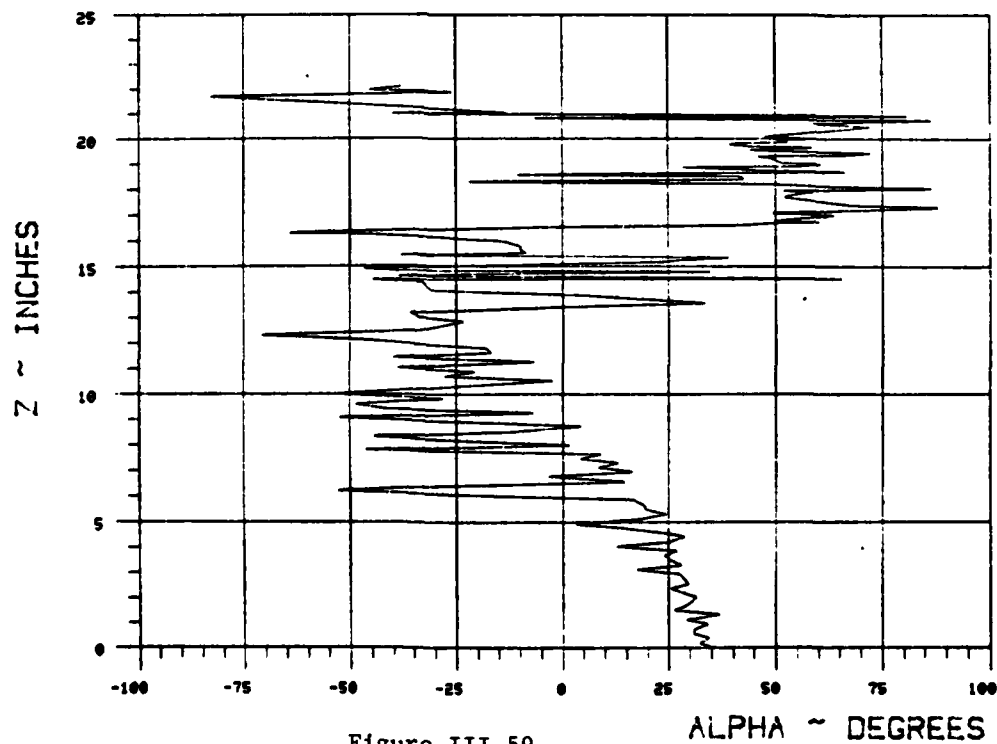
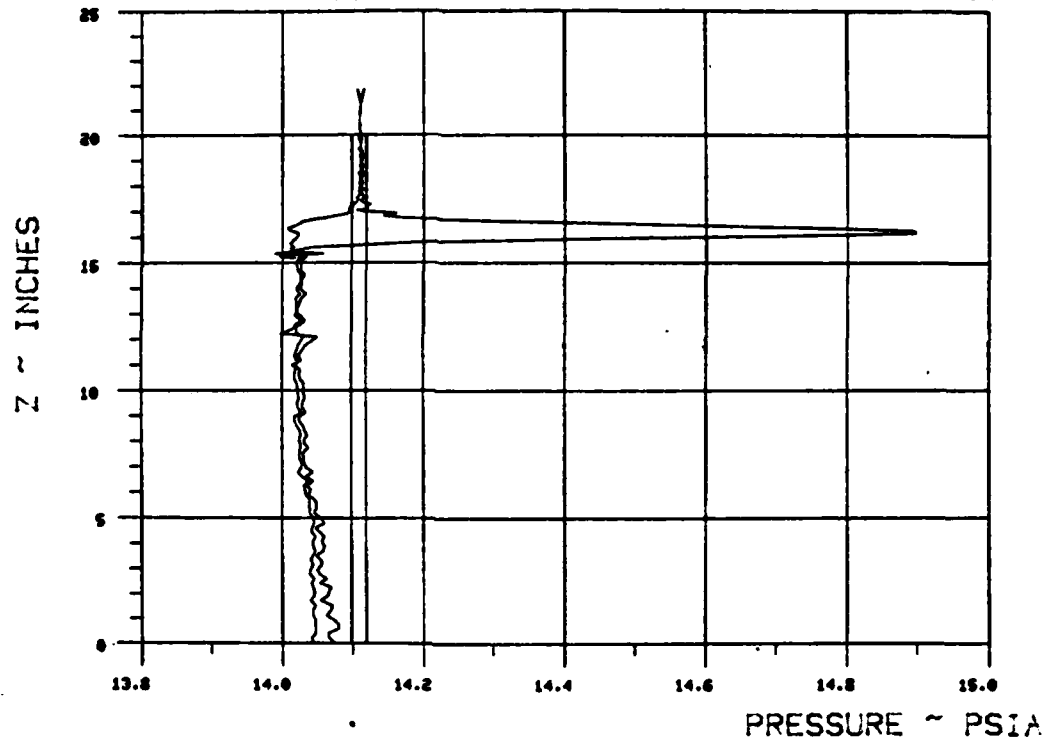


Figure III-59.

WAKE SURVEY DATA

TEST 287 RUN 17

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 7.4362



$X_{ss}/\text{chord} = \underline{1.899}$ $Y_{ss}/\text{chord} = \underline{.265}$

BETA = 70

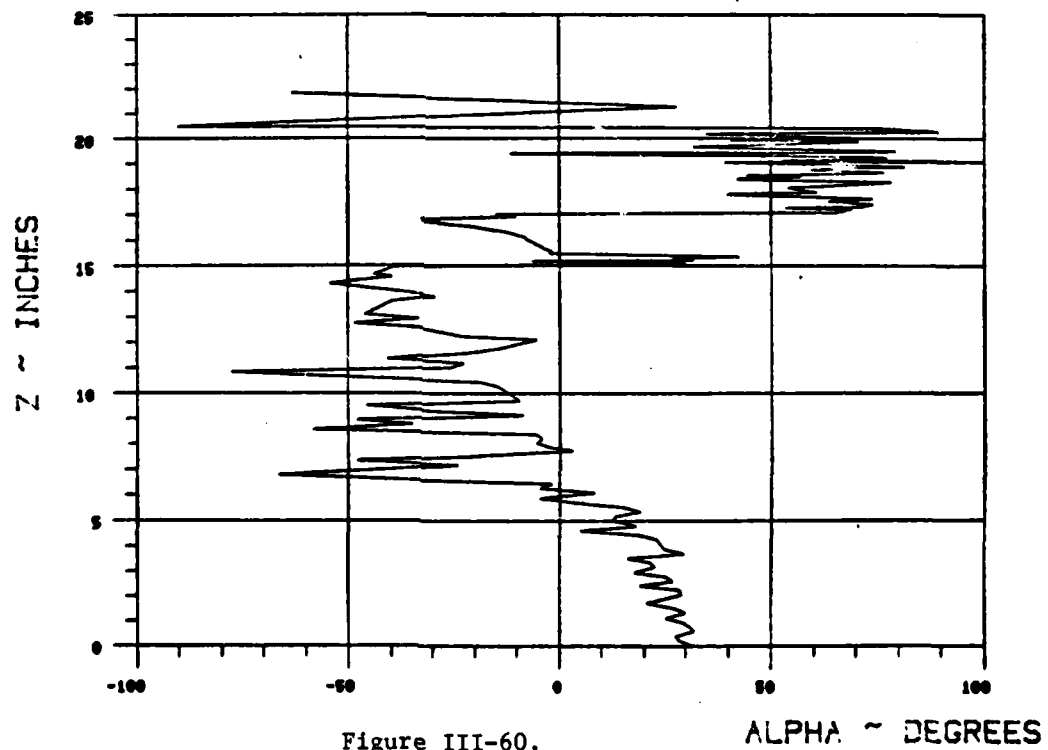
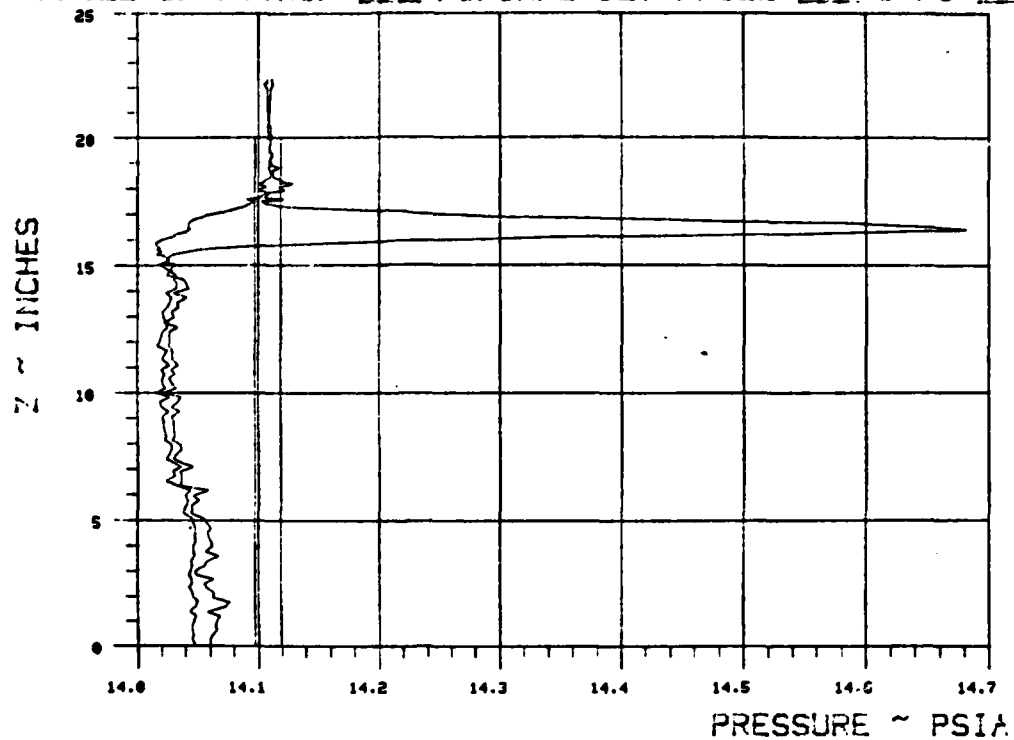


Figure III-60.

WAKE SURVEY DATA

TEST 287 RUN 18

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 7.3527



$X_{ss}/\text{chord} = \underline{1.942}$ $Y_{ss}/\text{chord} = \underline{.147}$

BETA = 70

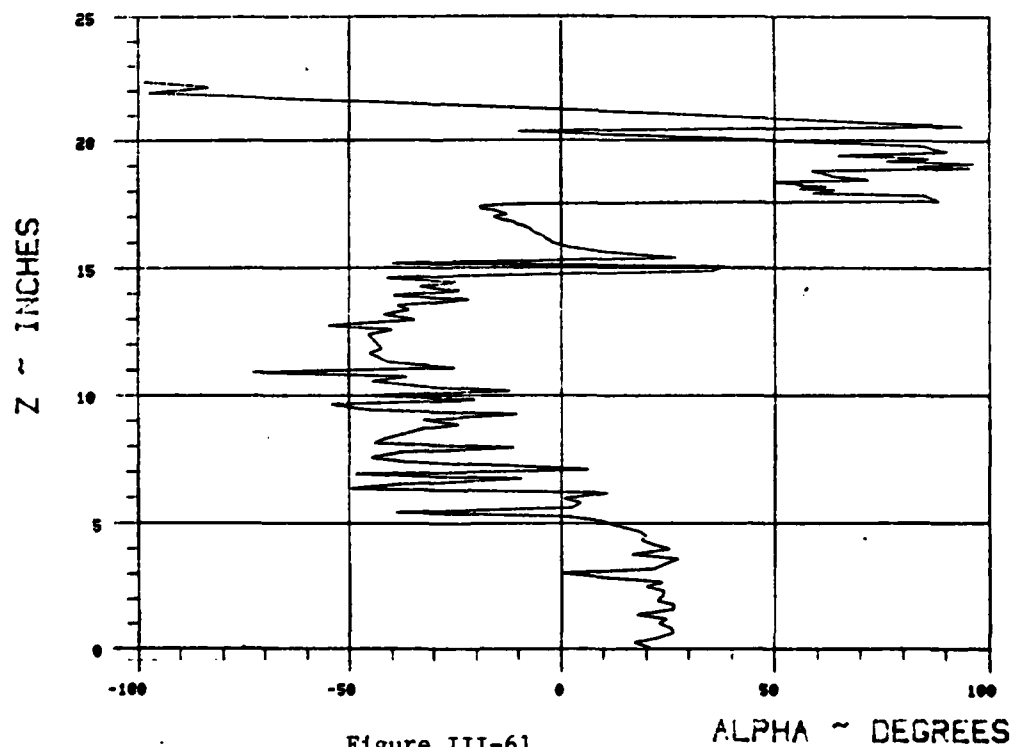
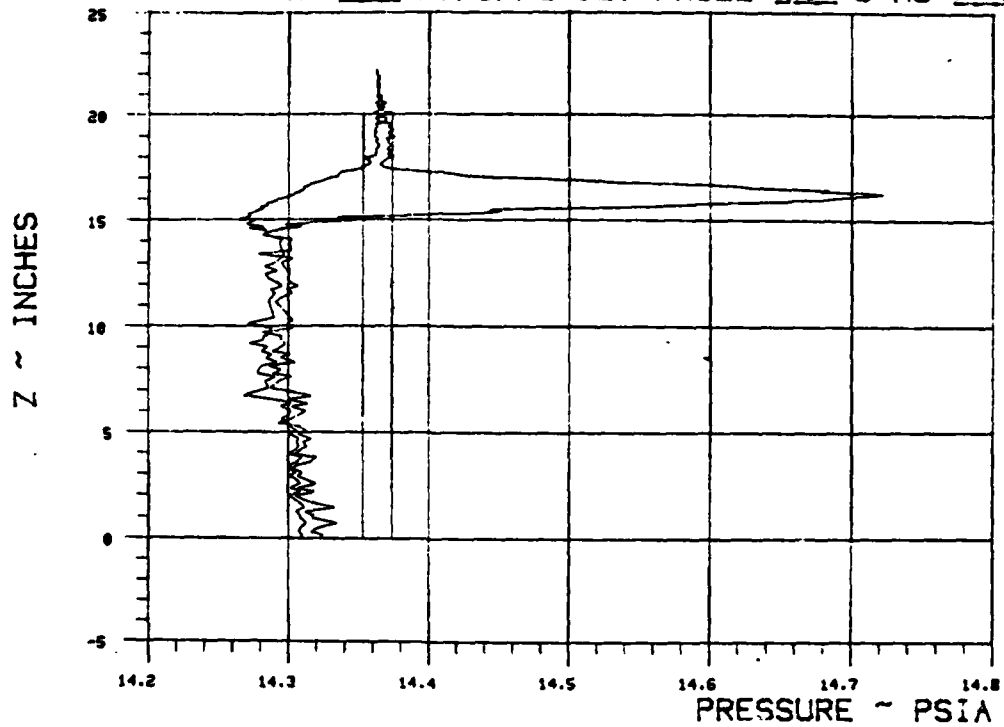


Figure III-61.

WAKE SURVEY DATA

TEST 287 RUN 19

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 7.7259



$X_{ss}/\text{chord} = 2.028$ $Y_{ss}/\text{chord} = -0.088$

BETA = 70

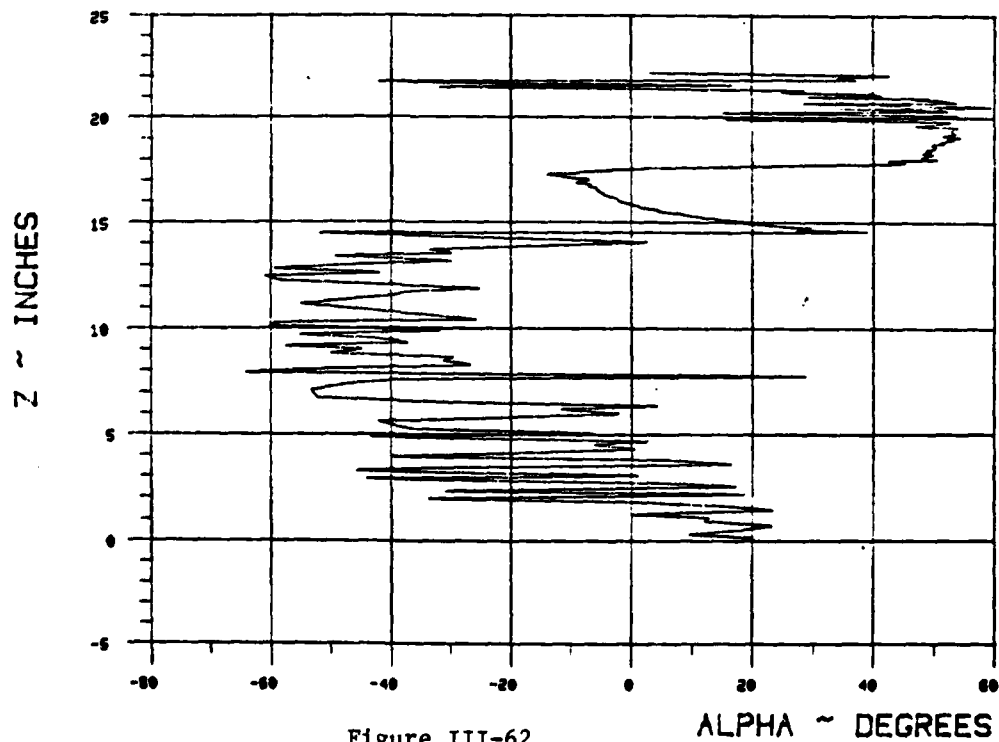
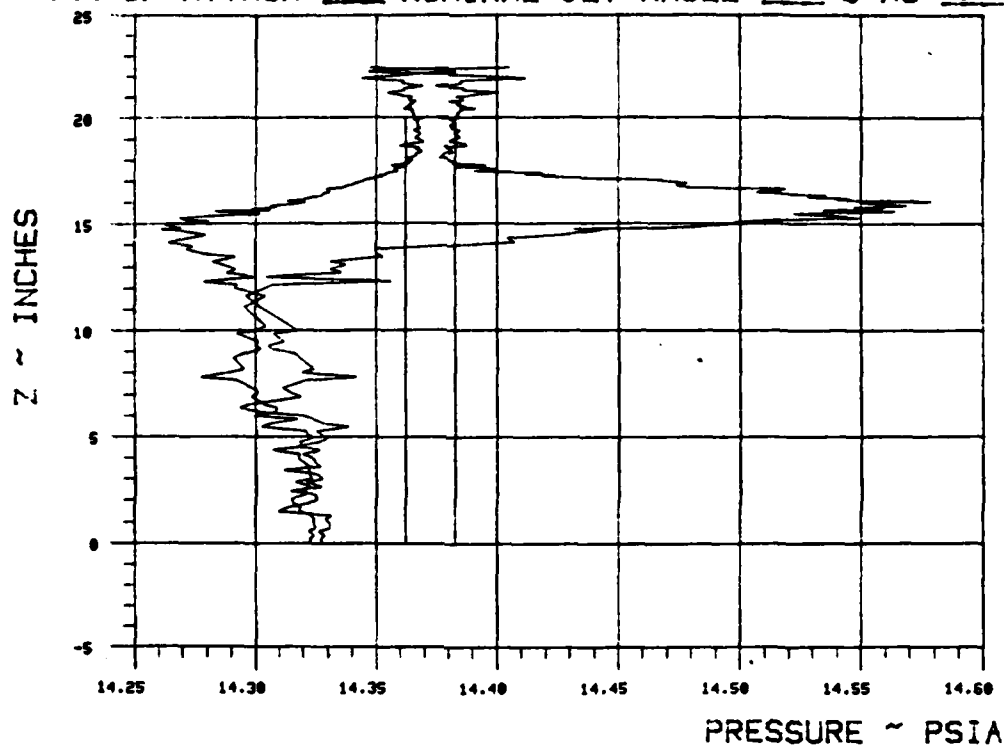


Figure III-62.

WAKE SURVEY DATA

TEST 287 RUN 20

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 7.6749



$X_{ss}/\text{chord} = \underline{2.156}$ $Y_{ss}/\text{chord} = \underline{-1.44}$

BETA = 70

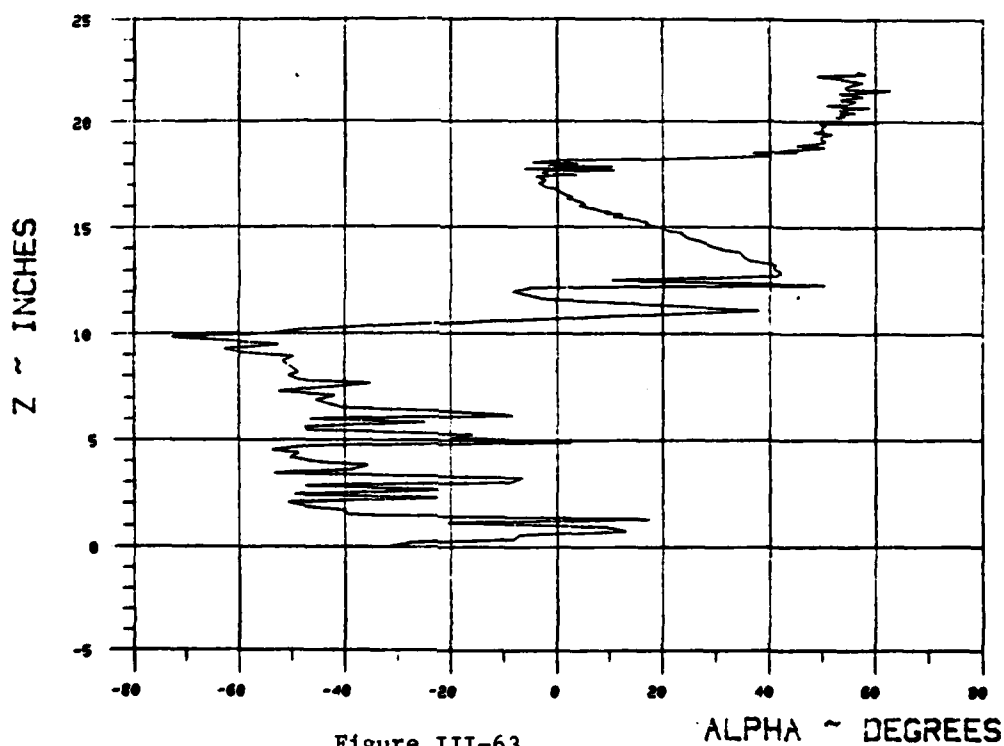
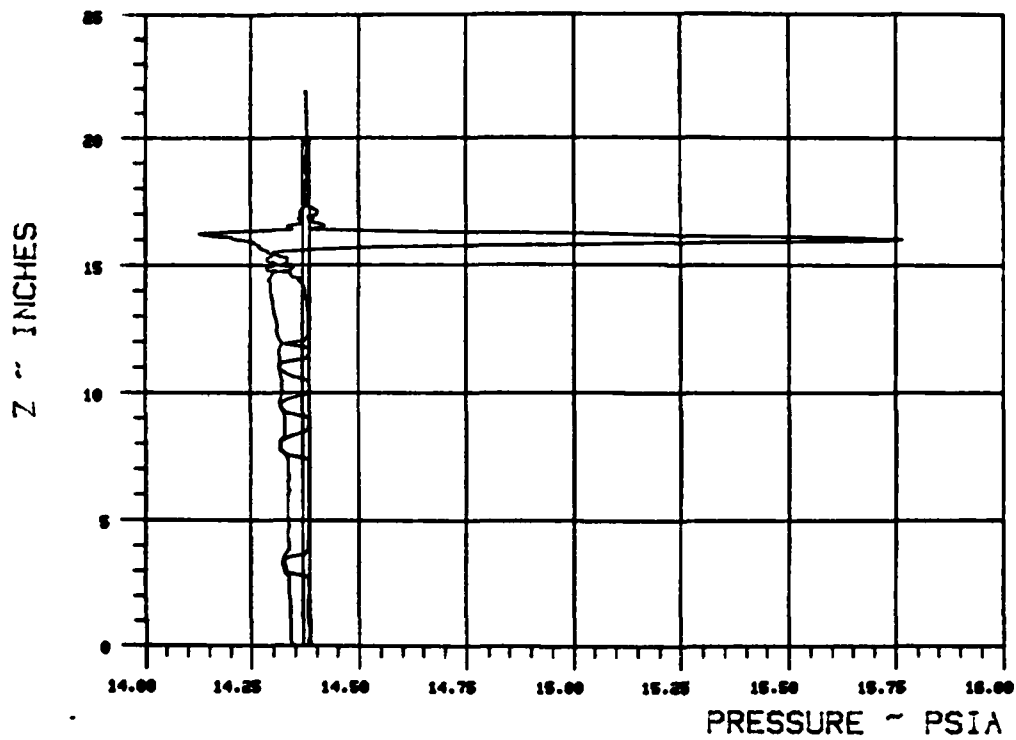


Figure III-63.

WAKE SURVEY DATA

TEST 287 RUN 21

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 11.4013



$X_{ee}/\text{chord} = \underline{1.852}$ $Y_{ee}/\text{chord} = \underline{.382}$

BETA = 70

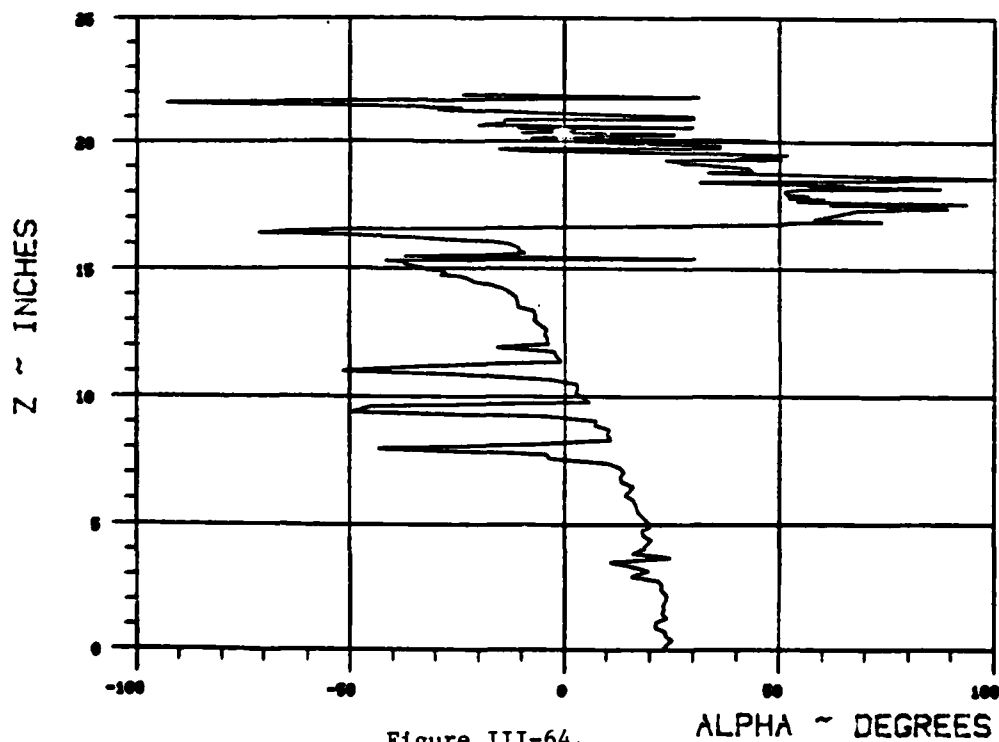
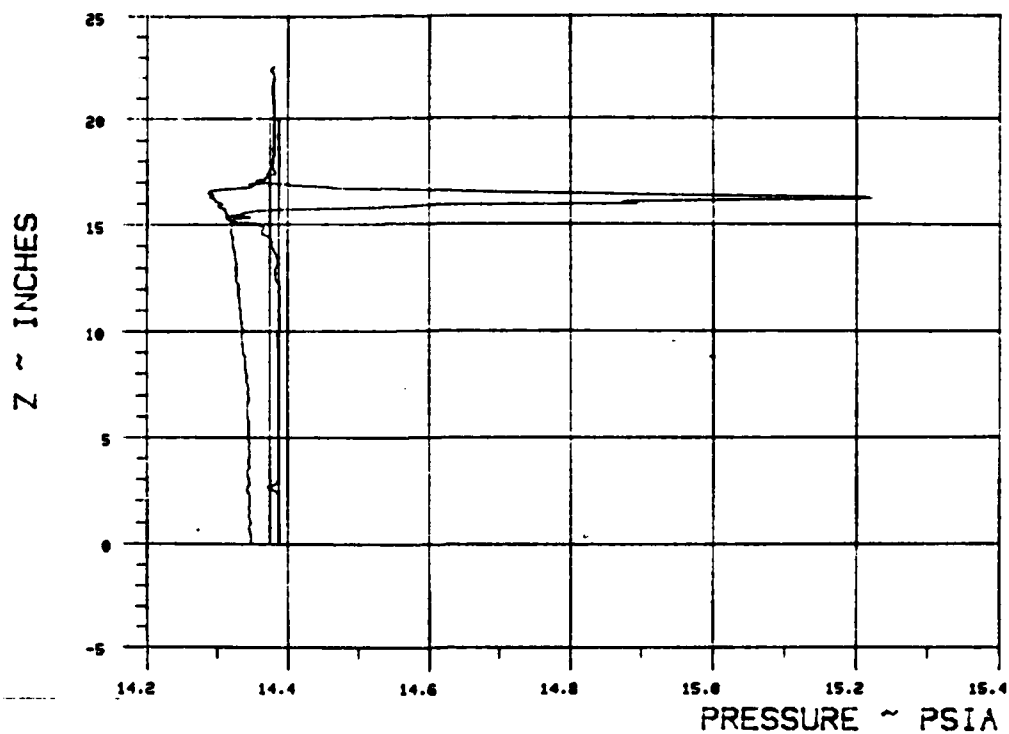


Figure III-64.

WAKE SURVEY DATA

TEST 287 RUN 22

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 11.4289



$X_{ee}/\text{chord} = \underline{1.899}$ $Y_{ee}/\text{chord} = \underline{.265}$

BETA = 70

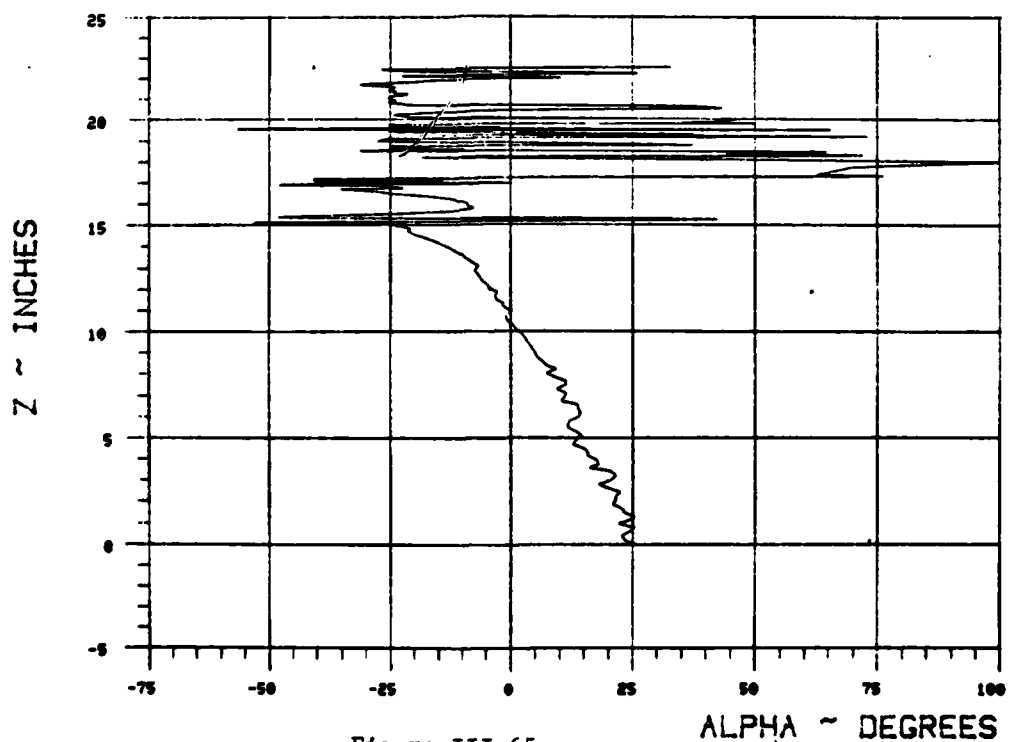
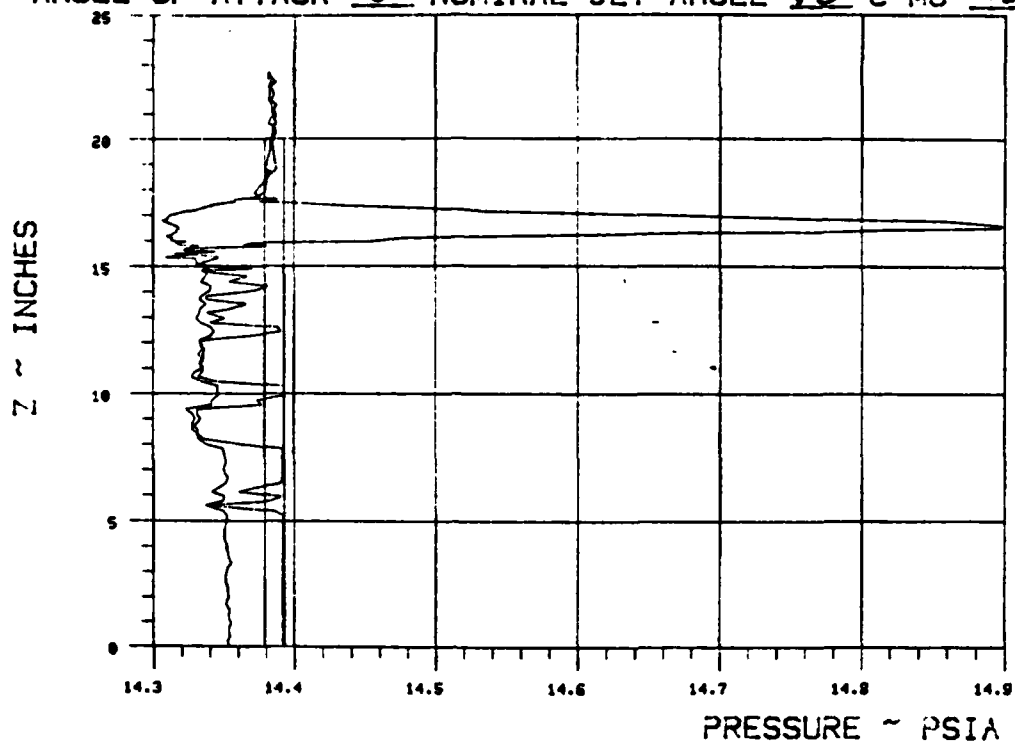


Figure III-65.

WAKE SURVEY DATA

TEST 287 RUN 23

ANGLE OF ATTACK 6 NOMINAL JET ANGLE 90 C MU 11.2001



$X_{ss}/\text{chord} = 1.942$ $Y_{ss}/\text{chord} = .147$

BETA = 20

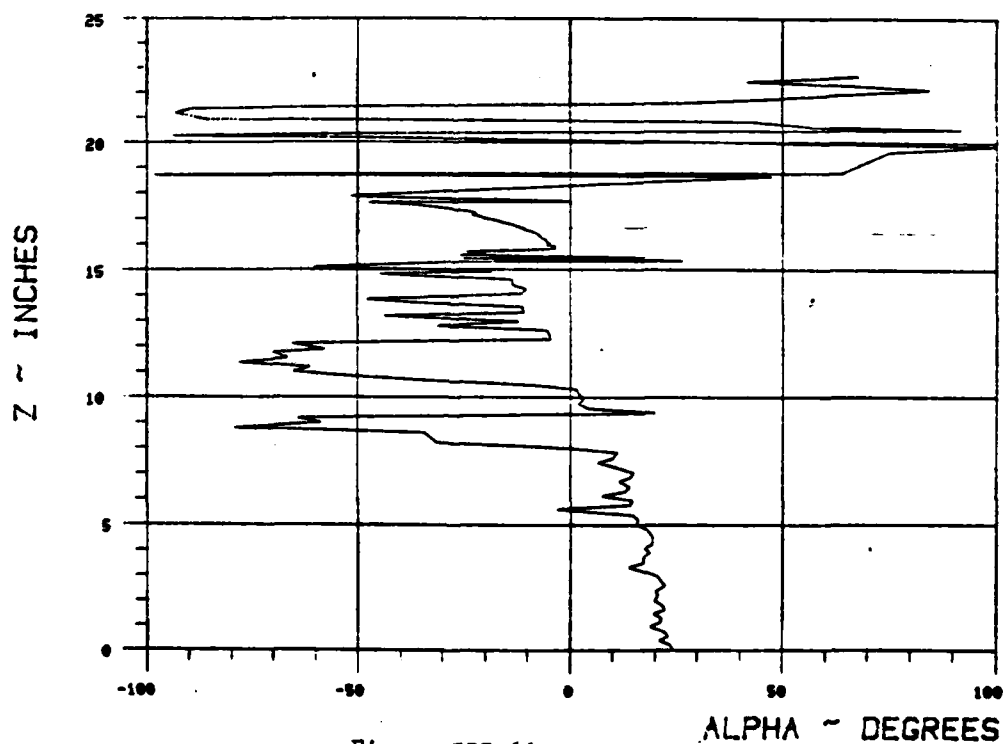
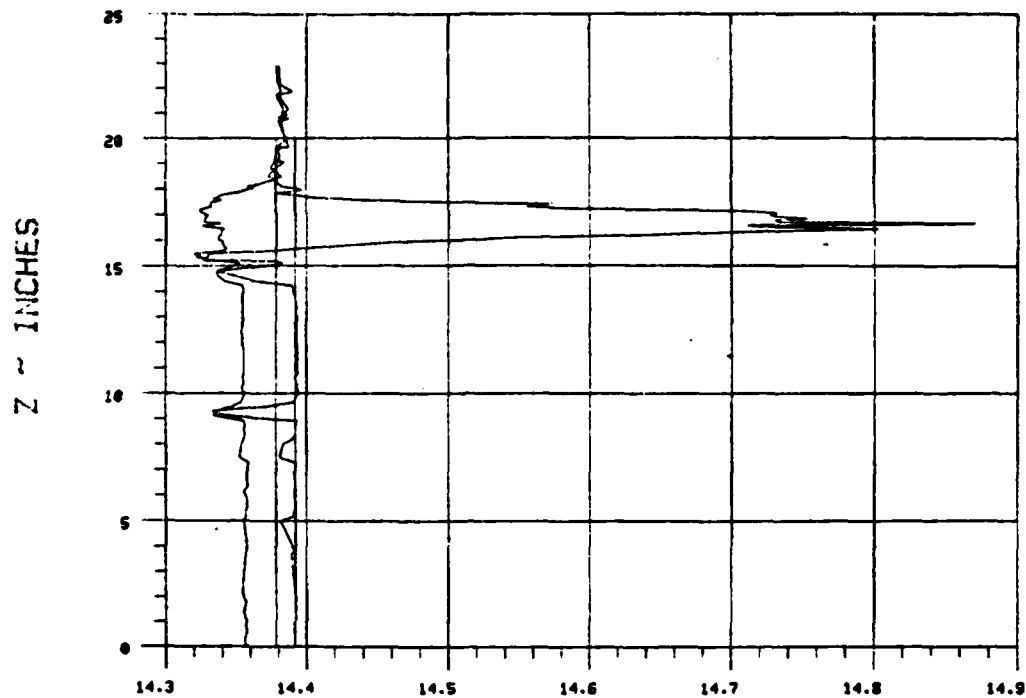


Figure III-66.

WAKE SURVEY DATA

TEST 287 RUN 24

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 11/33



PRESSURE ~ PSIA

$X_{90}/\text{chord} = 2.028$ $Y_{90}/\text{chord} = -0.088$

BETA = 70

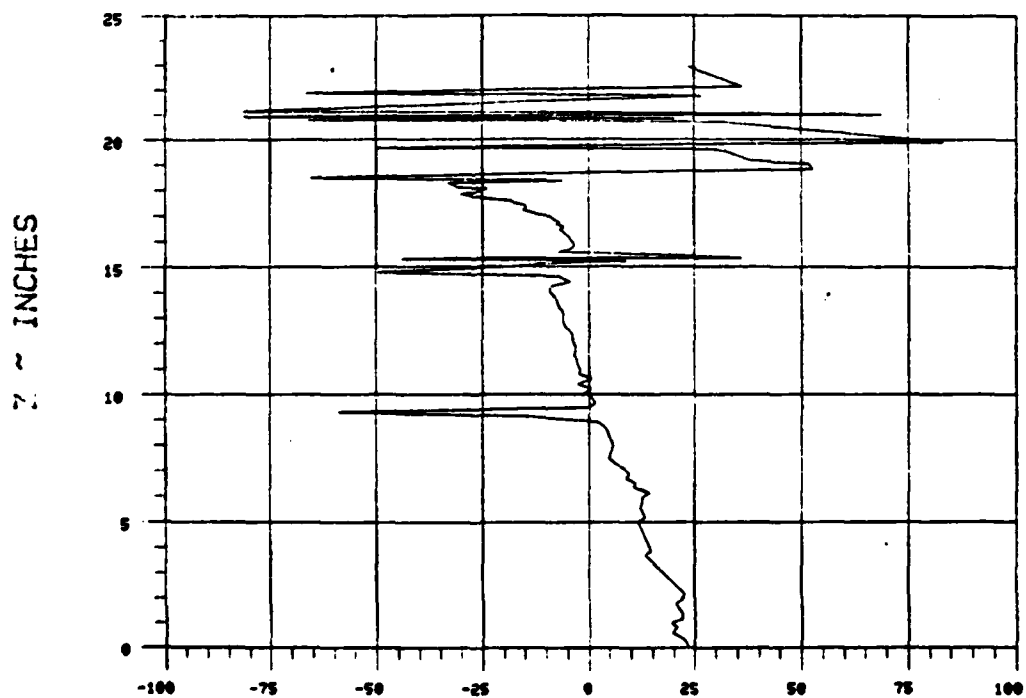


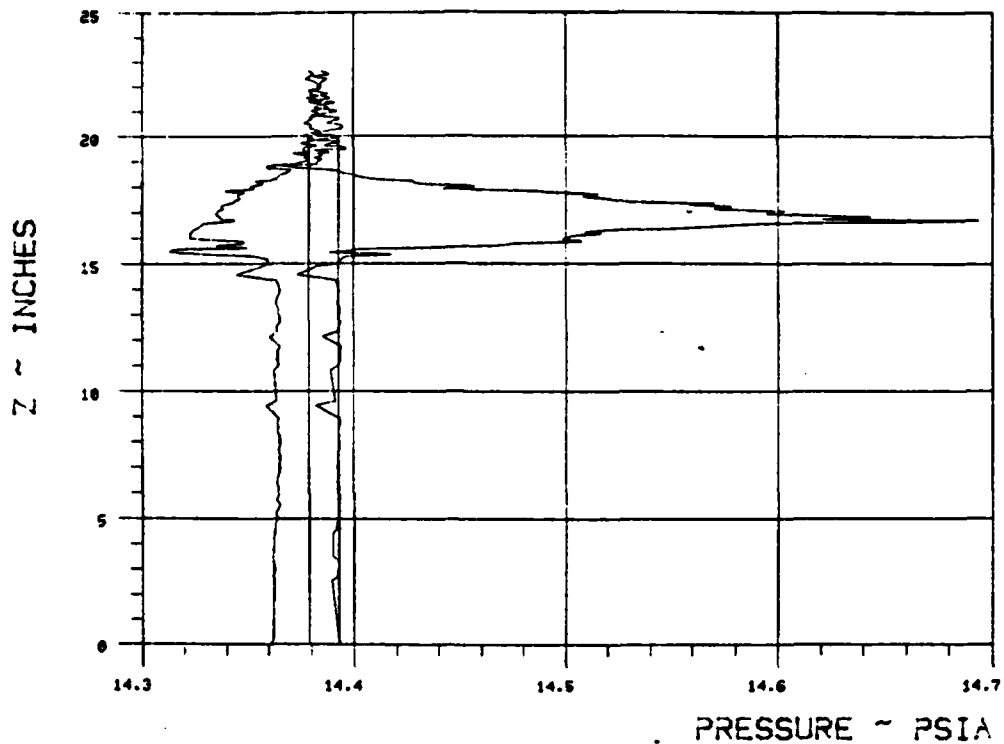
Figure III-67.

ALPHA ~ DEGREES

WAKE SURVEY DATA

TEST 207 RUN 25

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 11.4093



$X_{ss}/\text{chord} = 2.156$ $Y_{ss}/\text{chord} = 1.44$

BETA = 70

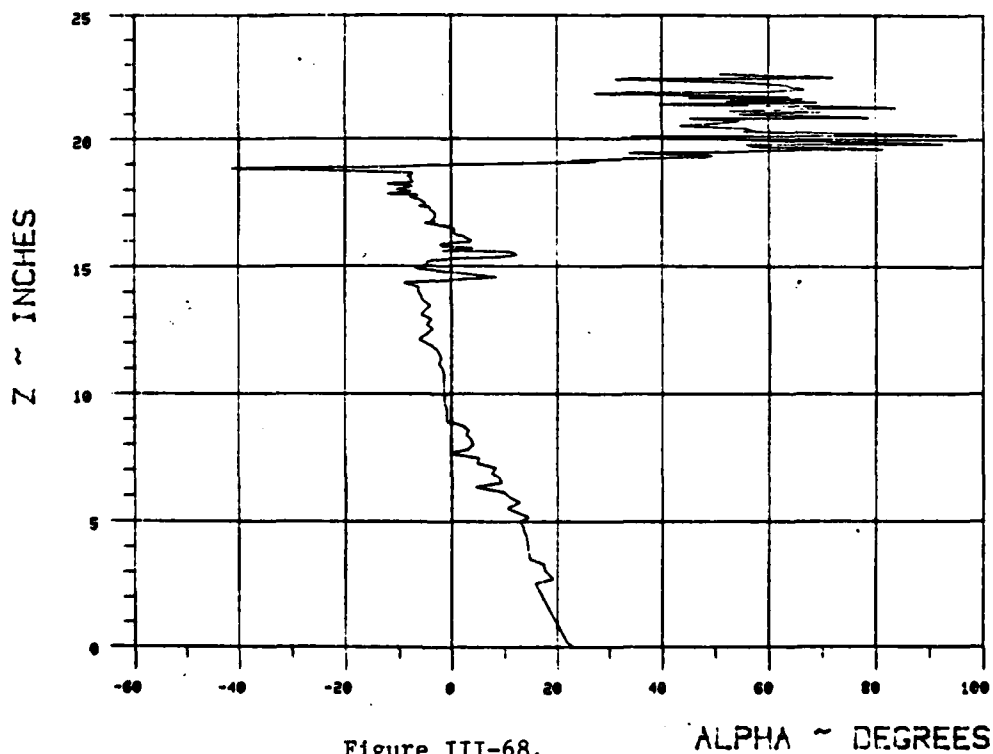


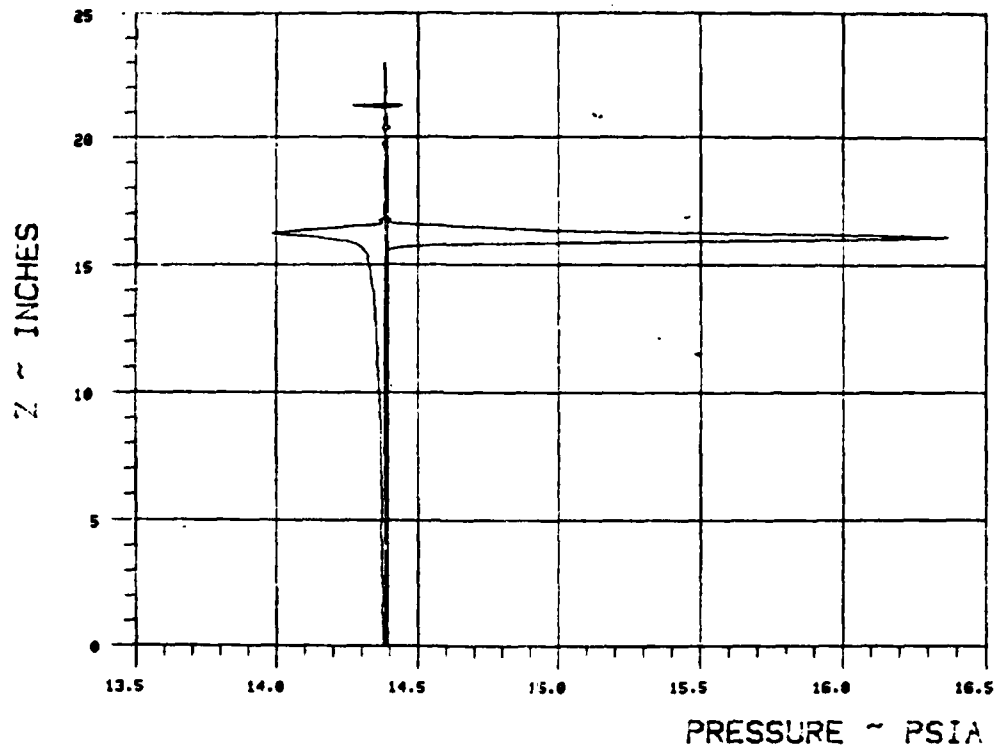
Figure III-68.

ALPHA ~ DEGREES

WAKE SURVEY DATA

TEST 287 RUN 26

ANGLE OF ATTACK 0 NOMINAL SET ANGLE 90 C MU 14.3862



$X_{ss}/\text{chord} = \underline{1.857}$ $Y_{ss}/\text{chord} = \underline{.382}$

BETA = 70

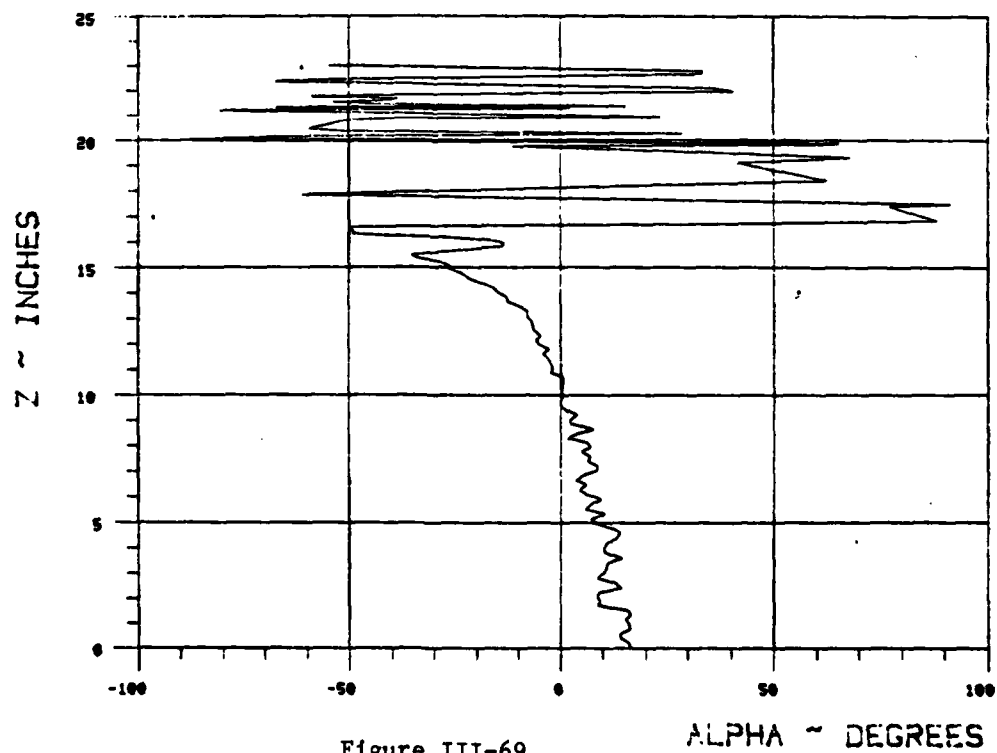
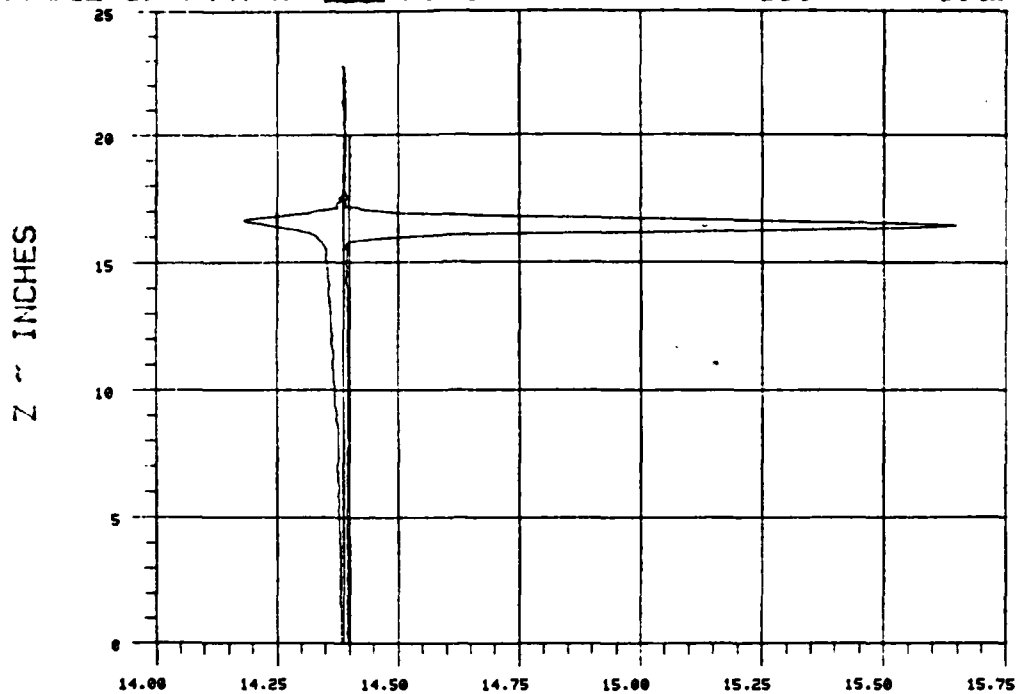


Figure III-69.

WAKE SURVEY DATA

TEST 207 RUN 27

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 14.4717



PRESSURE ~ PSIA

$X_{ss}/\text{chord} = \underline{1.899}$ $Y_{ss}/\text{chord} = \underline{.265}$

BETA = 70

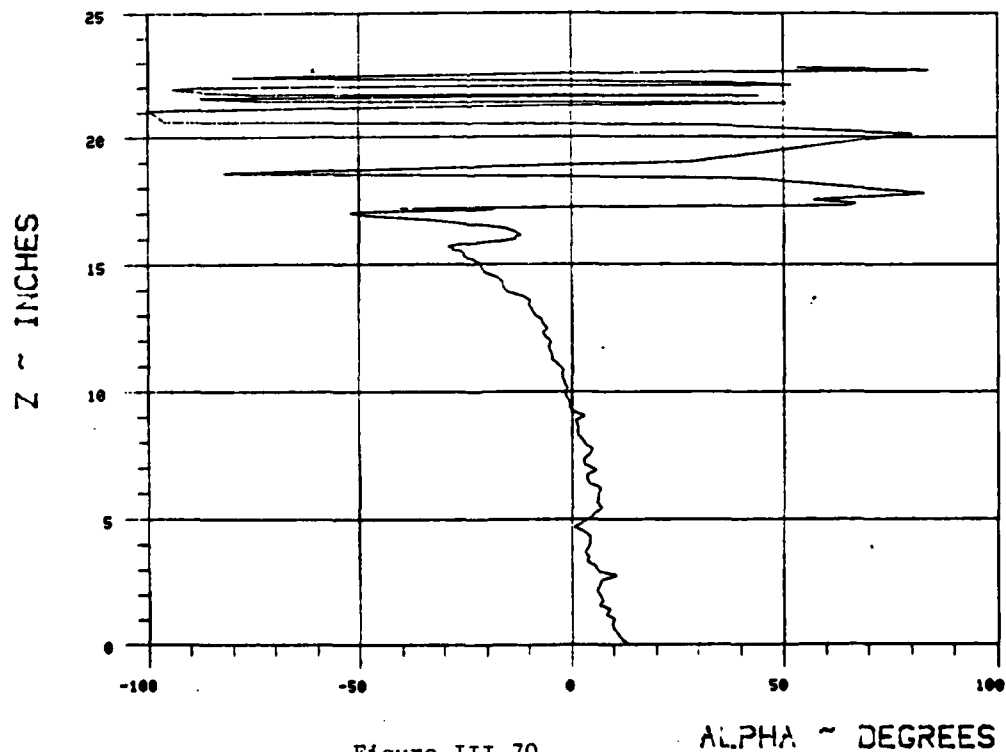


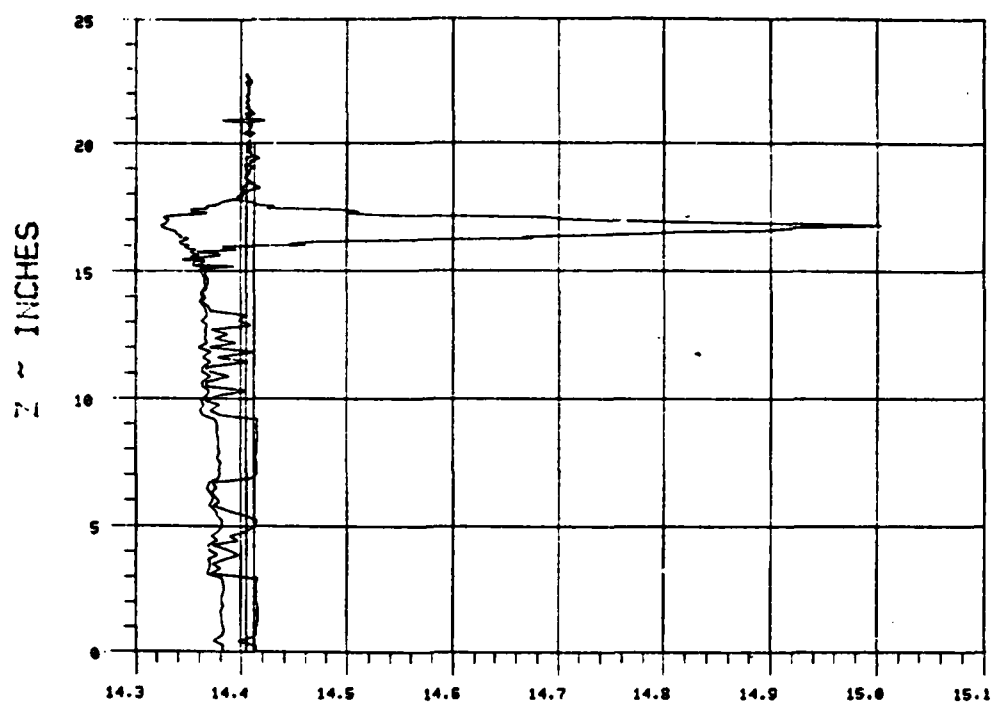
Figure III-70.

ALPHA ~ DEGREES

WAKE SURVEY DATA

TEST 287 RUN 28

ANGLE OF ATTACK 0 NOMINAL JET ANGLE 90 C MU 20.9048



PRESSURE ~ PSIA

$X_{ss}/chord = 1.942$ $Y_{ss}/chord = 1.47$

BETA = 70

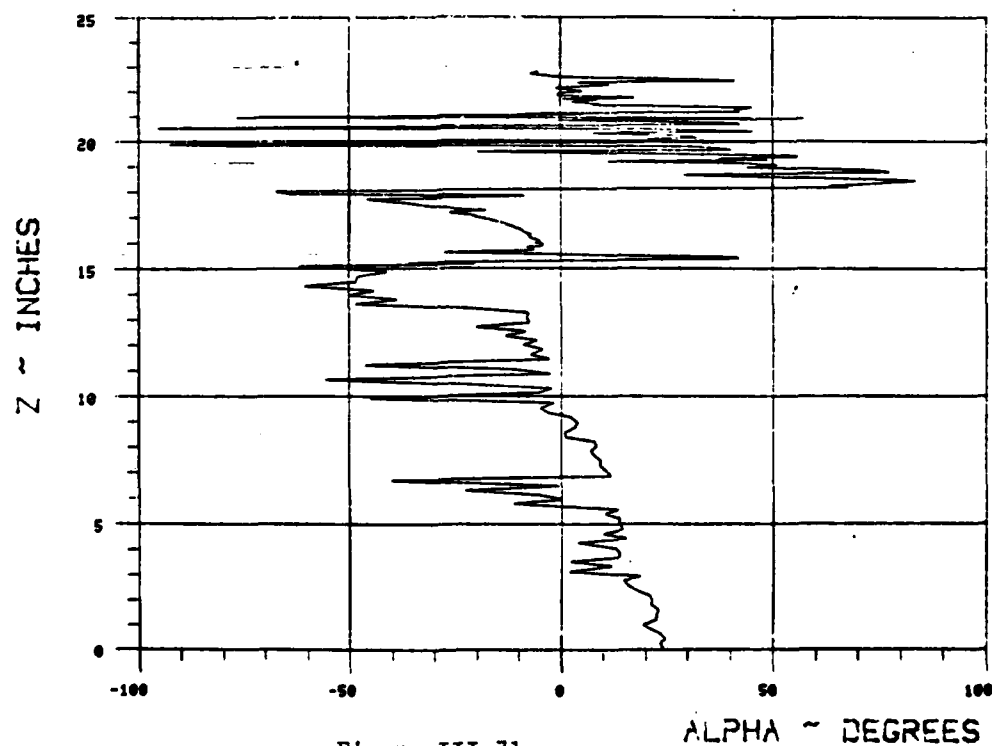


Figure III-71.

ALPHA ~ DEGREES

DATE
FILMED
8